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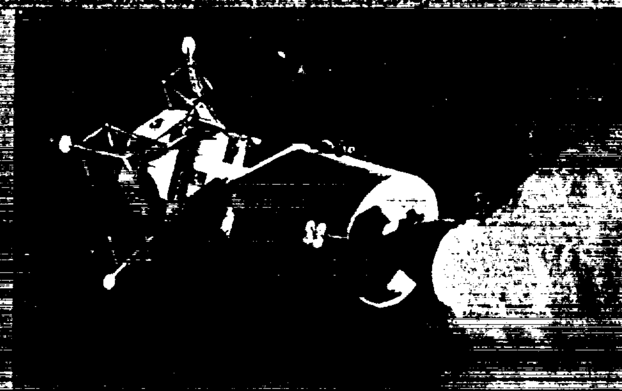
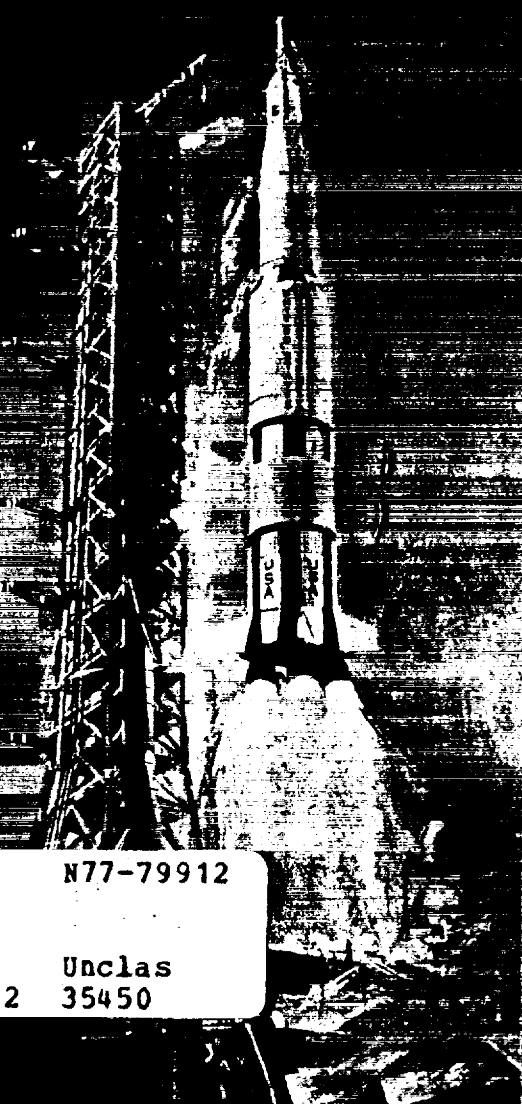


APOLLO SPACECRAFT FAMILIARIZATION



NASA SUPPORT MANUAL

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APOLLO
SPACECRAFT FAMILIARIZATION

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SITE SUPPORT DOCUMENTATION—DEPT 677

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LIST OF EFFECTIVE PAGES

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TECHNICAL REPORT INDEX/ABSTRACT

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DESCRIPTIVE TERMS

Section titles are Project Apollo, Apollo Space Vehicle, Spacecraft Systems, Lunar Excursion Module, Apollo Spacecraft Manufacturing, Apollo Training Equipment, Apollo Test Program, Lunar Landing Mission, Apollo Support Manuals, and Glossary of Abbreviations, Symbols, and Terms.

ABSTRACT

This issue of the Apollo Spacecraft Familiarization manual provides introductory data for personnel associated with the Apollo program. Each command and service module system is discussed in general terms, but with sufficient detail to convey a clear understanding of the systems. In addition, the Apollo earth orbit and lunar landing missions are described, planned, completed, and test programs or missions are identified. Manufacturing, training equipment, ground support equipment, space vehicles, and the lunar excursion module are all covered in gross terms.

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INTRODUCTION

This manual provides general introductory data for personnel associated with the Apollo program. Each command and service module system is discussed in general terms, but with sufficient detail to convey a clear understanding of the systems. In addition, the Apollo earth orbit and lunar landing missions are described, planned, completed, and test programs or missions are identified. Manufacturing, training equipment, ground support equipment, space vehicles, and the lunar excursion module are all covered in gross terms. The source information used in the preparation of this manual was that available as of November 1, 1965.

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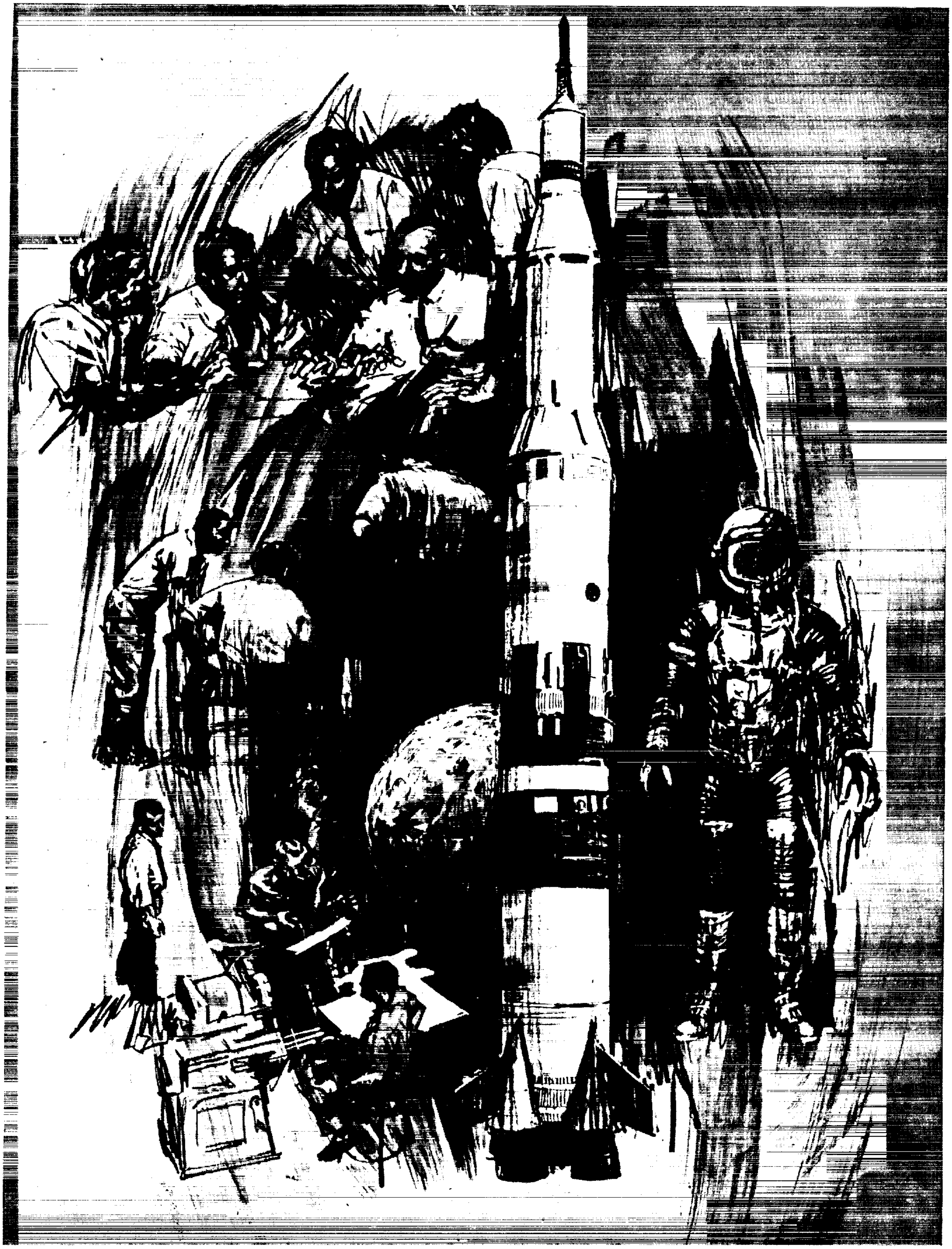

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PROJECT APOLLO

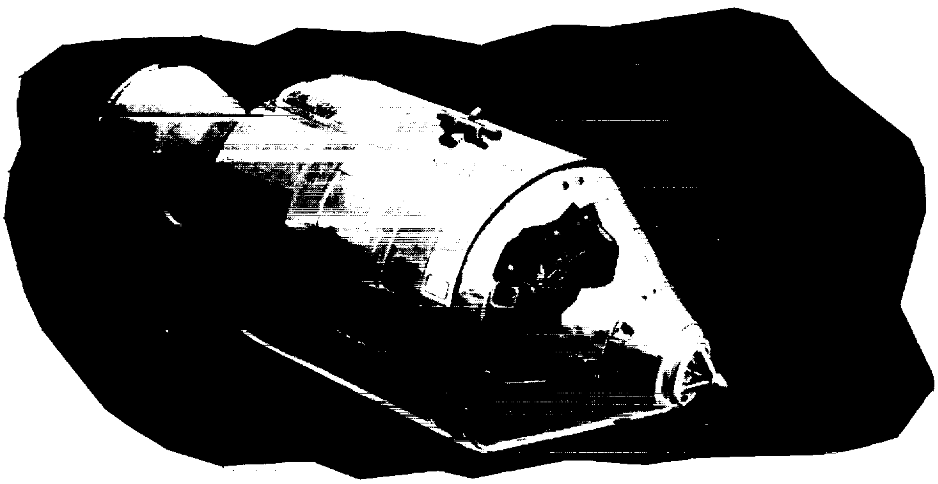

Figure 1-1. Apollo Spacecraft

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1-1. The ultimate objective of Project Apollo is to land men on the moon for limited observation and exploration in the vicinity of the landing area and assure their subsequent safe return to earth. (See figure 1-1.) This objective will climax a series of earth suborbital and orbital missions. Although each of these missions will have specific objectives, they will be flown primarily for state-of-the-art advancement and qualification of systems for the ultimate lunar landing mission.

1-2. The project consists of three phases designed to obtain these ultimate goals:

- The first phase consists of a number of boilerplate missions for research and developmental purposes. Boilerplates are preproduction spacecraft similar to their production counterparts in shape, size, mass, and center of gravity.
- Phase two will be conducted with limited production spacecraft. These spacecraft will be utilized for systems development and qualification in an earth orbital environment with man in the loop and without the lunar excursion module.
- The third and final phase consists of those missions with the lunar excursion module which will culminate in a manned lunar landing.

1-3. THE APOLLO TEST PROGRAM.

1-4. The Apollo test program is designed to confirm the overall structural integrity, systems performance, compatibility, and intermodular compatibility of the three-man spacecraft. The program follows a path of developmental progress from initial structural integrity confirmation to the complex testing of each module and system for reliability and compatibility. Three basic phases are scheduled for spacecraft testing. The first is research and developmental testing conducted to verify the engineering concepts and basic design employed in the Apollo configuration. The second phase is the qualification testing of the spacecraft hardware and components. The third phase of the test program will verify the production spacecraft systems operation and the man-machine compatibility of the spacecraft. A full test program summary is presented in section VII.

1-5. EARTH SUBORBITAL MISSIONS.

1-6. Unmanned spacecraft earth suborbital missions are scheduled to evaluate the command module heat shield performance, and the structural compatibility and integrity of the spacecraft and launch vehicle. These missions will also serve to qualify and confirm compatibility of the spacecraft-launch vehicle combinations.

1-7. The earth suborbital missions will aid in the determination of structural loading, systems performance, and separation characteristics of the launch escape system and boost protective cover from the command module, service module from the adapter, and the command module from the service module. Also, the command module adequacy for manned entry from a low earth orbit will be determined as well as performance of the service module reaction control system ullage maneuver, service propulsion start, and service propulsion system operation.

1-8. An example of a mission profile for one particular earth suborbital mission is presented in figure 1-2.

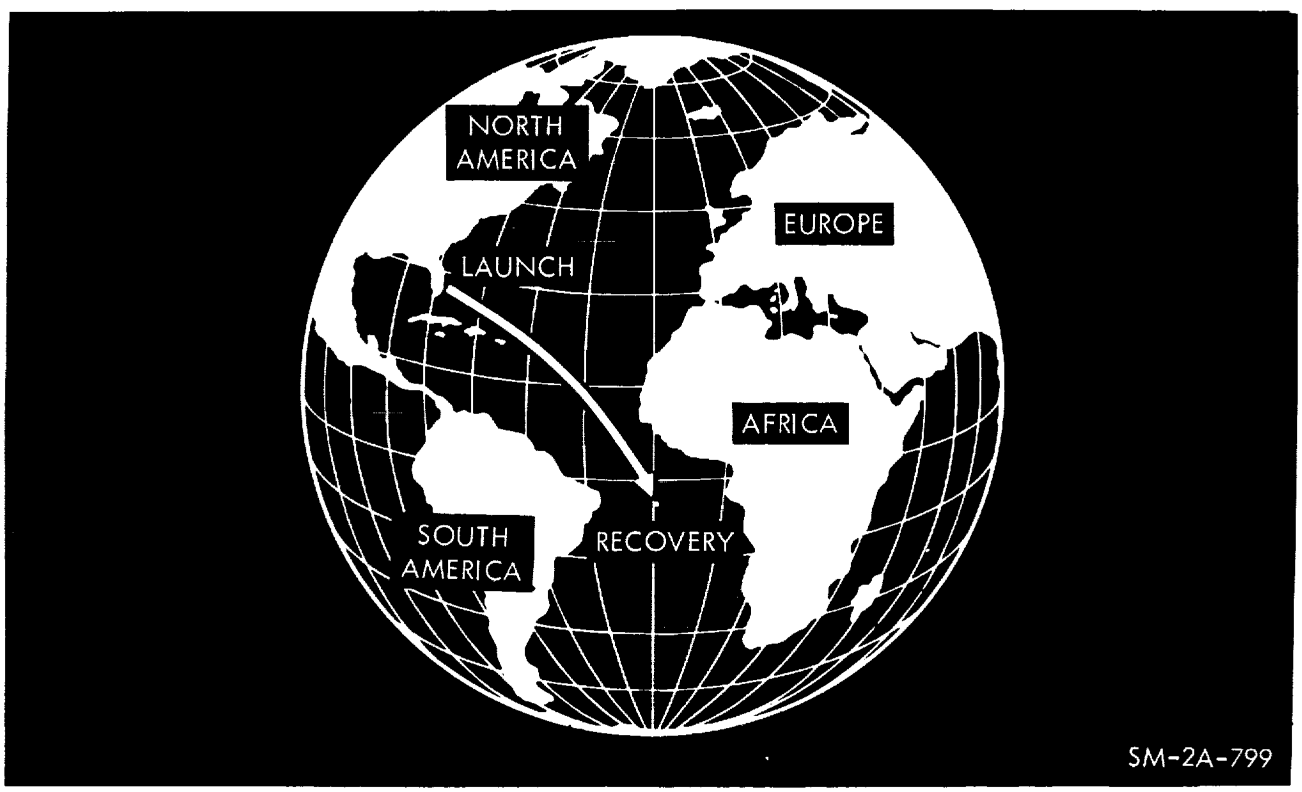


Figure 1-2. Earth Suborbital Mission Profile (Typical)

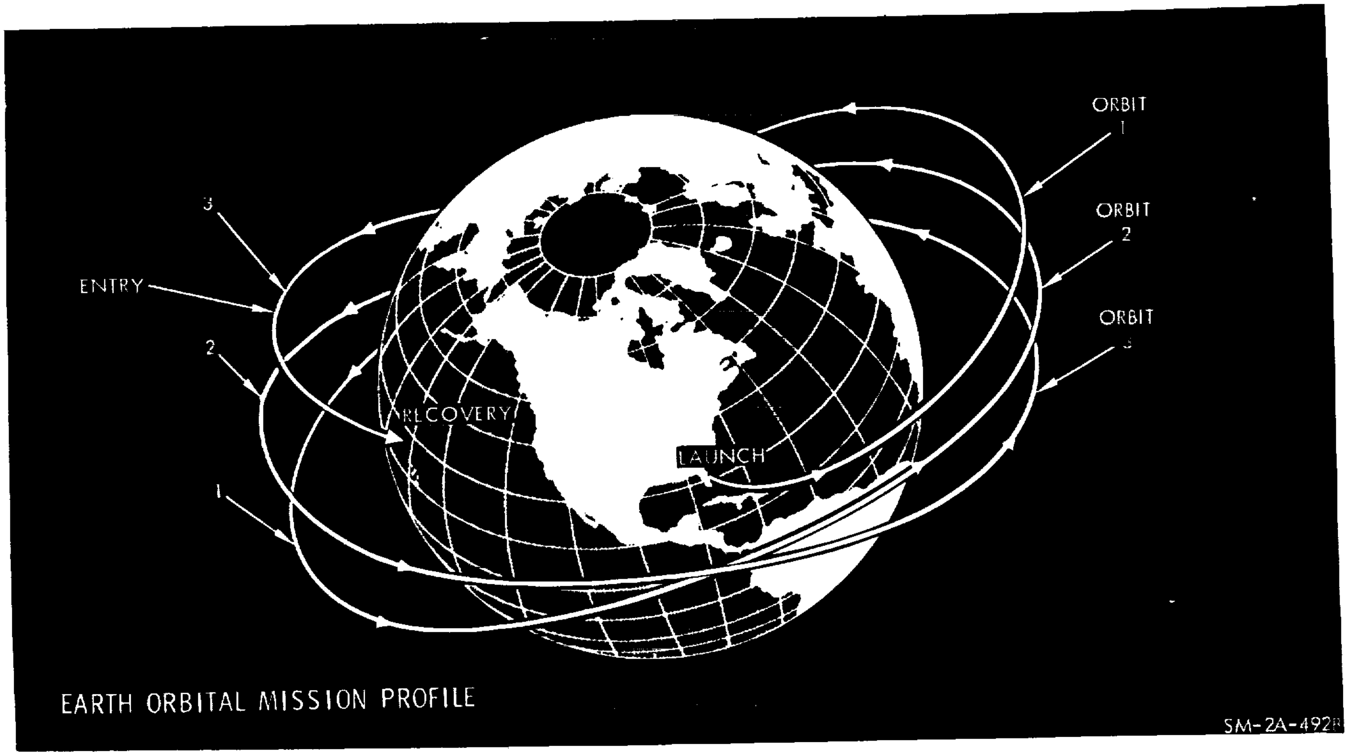


Figure 1-3. Earth Orbital Mission Profile (Typical)

1-9. EARTH ORBITAL MISSIONS.

1-10. Unmanned and manned spacecraft earth orbital missions (figure 1-3) are programmed to confirm the compatibility of the spacecraft-launch vehicle and spacecraft-LEM combinations to demonstrate spacecraft and lunar excursion module (LEM) performance, and to develop flight crew proficiency.

1-11. The unmanned missions will serve to qualify the launch vehicle and confirm spacecraft-launch vehicle compatibility. Operating procedures will be analyzed during these missions to determine the adequacy, feasibility, and overall performance of the launch vehicle and the unmanned spacecraft.

1-12. Manned missions will be conducted to determine crew and manned space-flight network (MSFN) proficiency in ascent, earth orbit injection, transposition and docking, rendezvous and docking, entry, and recovery-phase task requirements. Flight crew MSFN interface will be constantly conditioned and prepared for deep-space operations.

1-13. The individual mission profile will be determined by the mission objectives for a given flight. The profiles of earth orbital missions range from circular orbits to elliptical orbits.

1-14. LUNAR LANDING MISSION.

1-15. The lunar landing mission will be accomplished after all other tests and missions have been satisfactorily completed. The purpose of this mission is to explore the lunar surface in the vicinity of the LEM, and to evaluate the effect of the deep-space environment upon the flight crew, spacecraft, and the MSFN.

1-16. The lunar landing mission will be of much greater complexity than previous mission missions. In addition to those tasks required for an earth orbital mission, translunar injection, translunar midcourse corrections, lunar orbit insertion and coast, LEM descent, lunar exploration, LEM ascent, transearth injection, and transearth midcourse corrections must be accomplished.

1-17. The velocity required for the proper mission profile will be determined by MSFN and verified by the guidance computer of the spacecraft navigation and control system. After achieving lunar orbit, the flight crew will make observations of a preselected landing site to determine the adequacy of the landing area and/or possible alternate site. Two crew members will then enter the LEM through the forward tunnel of the command module, perform a check of the LEM systems, and extend the landing gear. At a predetermined point in lunar orbit, the LEM will separate from the command and service modules (CSM) and descend to the surface of the moon. The CSM, under control of the remaining crew member, will continue to orbit the moon.

1-18. After landing on the lunar surface, the LEM crewmen will egress to the lunar surface and explore the landing site area. During this time, samples of the lunar crust will be taken for subsequent analysis upon return to earth.

1-19. After lunar exploration has been completed, the crew members will re-enter the LEM, which will then ascend to rendezvous with the CSM. When docking is completed, the two LEM crew members enter the CSM, which is then separated from the LEM. Transearth injection (for return to earth), transearth midcourse corrections, entry, and recovery, will then be accomplished.

1-20. The navigational tasks required for the lunar landing mission far exceeds those of earth orbital missions. During this mission, the proficiency of flight crew navigation will undergo its severest test of the Apollo program. Figure 1-4 illustrates the typical lunar exploration mission profile with emphasis placed upon the major navigational tasks of the earth-moon relationship. The detailed requirements of the lunar landing (and exploration) mission are described in section VIII.

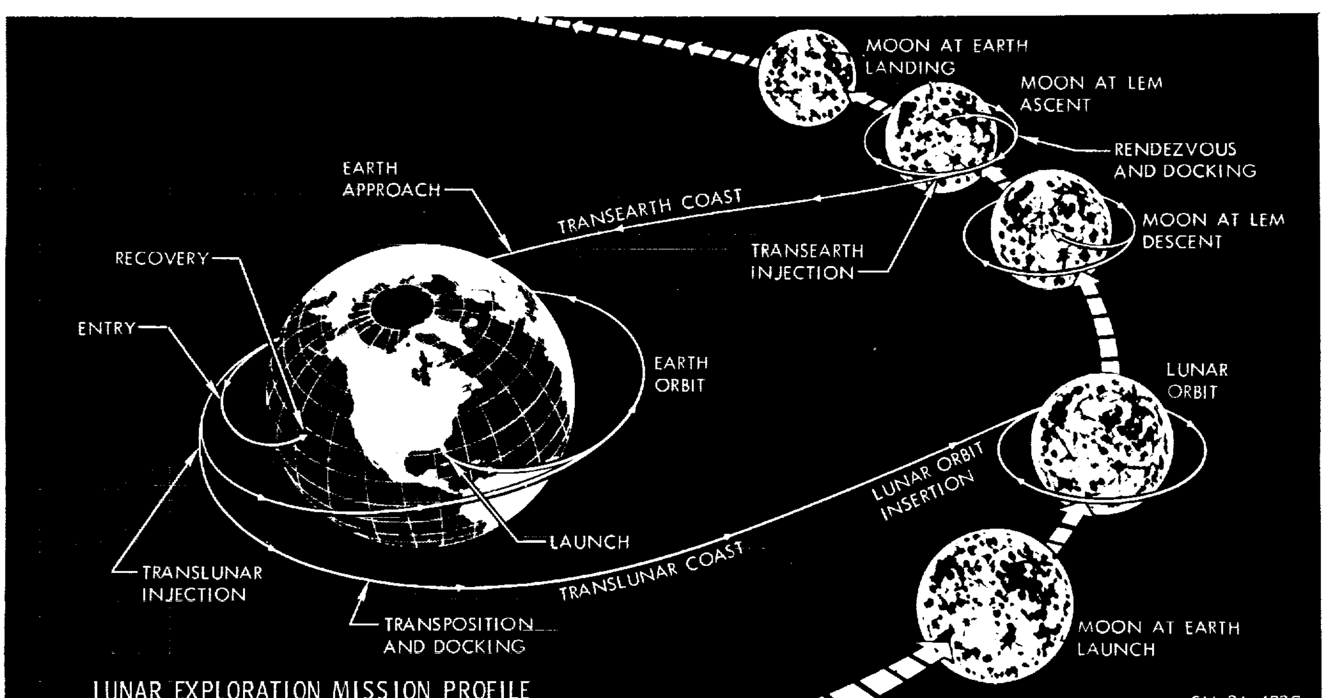


Figure 1-4. Lunar Exploration Mission Profile (Typical)

APOLLO SPACE VEHICLE

2-1. GENERAL.

2-2. The Apollo space vehicles are comprised of various spacecraft and launch vehicle modules. The spacecraft (at launch), based upon mission objectives, may consist of a launch escape system, command module, service module, spacecraft LEM adapter (SLA), and lunar excursion module. (See figure 2-1.) The launch vehicle consists of either a Saturn booster configuration or Little Joe II booster. The overall height and weight of the space vehicle is directly related to the flight trajectory dictated by mission objectives. Major variances in height and weight are based on the selection of the launch vehicle and configuration of the spacecraft.

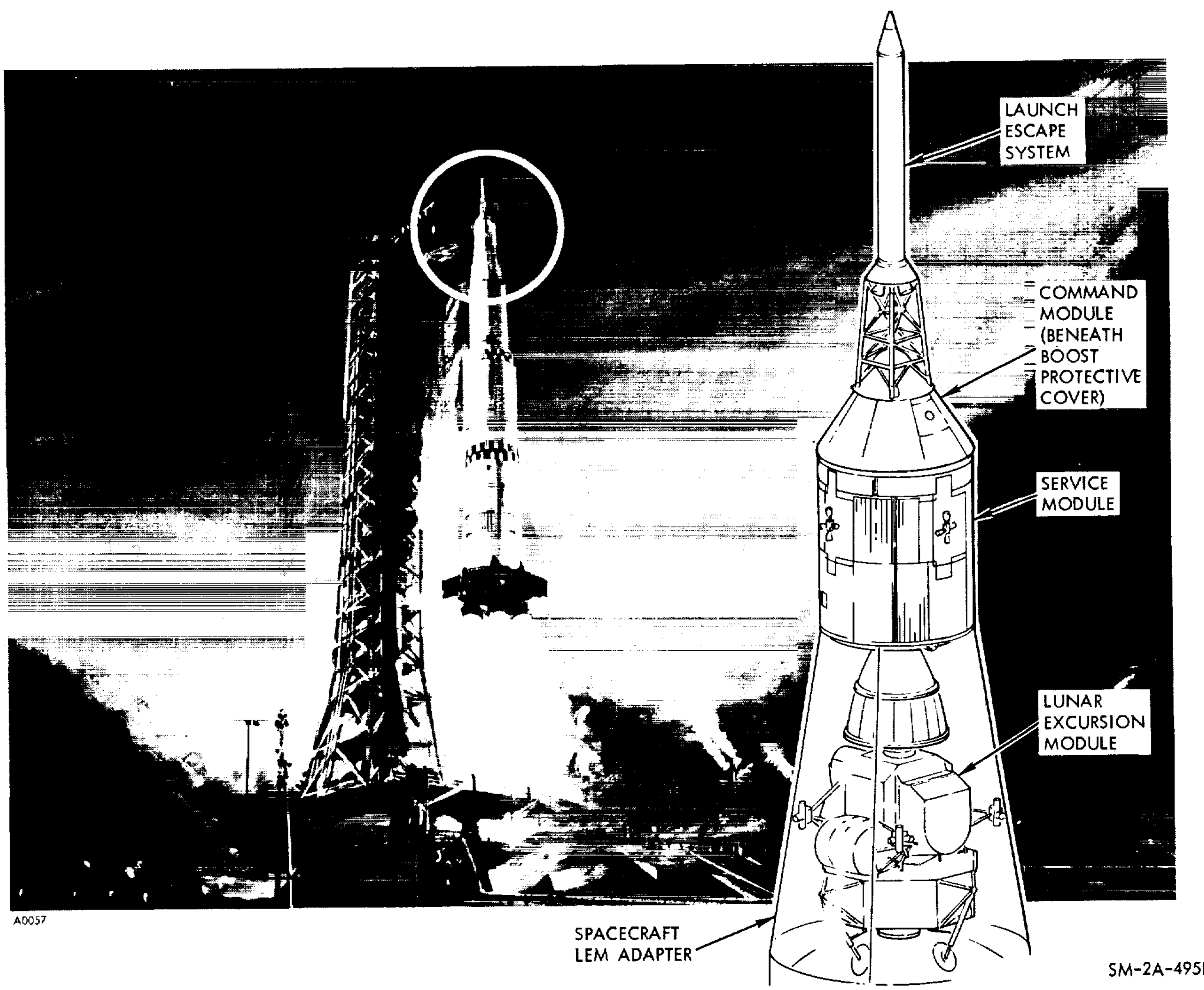


Figure 2-1. Apollo Space Vehicle

2-3. The external dimensions of the LES, C/M, S/M, and SLA remain constant. The LEM is housed within the SLA and will be installed in the space vehicle for the earth orbital rendezvous and docking missions, and the lunar landing missions.

2-4. Figure 2-1 depicts the lunar landing space vehicle configuration and geometry of the spacecraft. The launch vehicle configurations are illustrated later within this section. Refer to the description of launch vehicles for booster configuration variances.

2-5. APOLLO SPACECRAFT.

2-6. LAUNCH ESCAPE SYSTEM.

2-7. The launch escape system provides a means of removing the command module from the space vehicle during a pad abort or suborbital flight abort. The launch escape vehicle (figure 2-2) consists of a Q-ball (nose cone), ballast compartment, canard system, three rocket motors enclosed within an Inconel housing, a structural skirt, an open-frame tower, and a boost protective cover. The structural skirt is secured to the launch escape tower (LET), which transmits stress loads between the launch escape motor and the command module. The boost protective cover (BPC), which protects the C/M exterior during launch and boost, is fastened to the lower end of the tower. Four explosive bolts, one in each tower leg well, secure the tower to the command module structure. After a successful launch

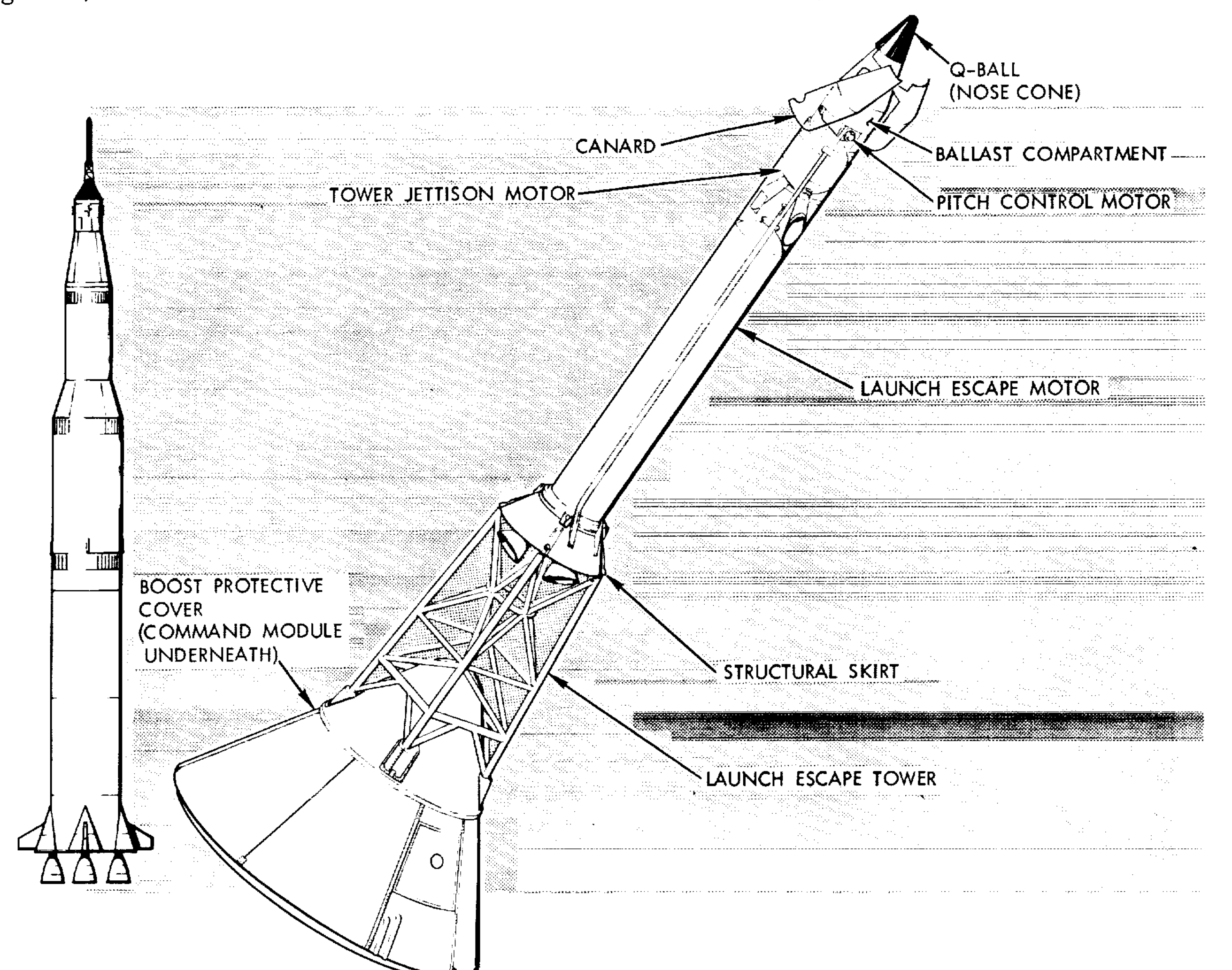


Figure 2-2. Launch Escape Vehicle

or abort mode initiation, explosive squibs ignite the explosive bolt charges which fracture the bolts and free the tower together with the boost protective cover. The rocket motors, canards, and explosive squibs are activated by electronic sequencing devices within the LES. Refer to section III for system operational data.

2-8. COMMAND MODULE.

2-9. The command module (figure 2-3) is the recoverable portion of the spacecraft and, houses the flight crew, the equipment necessary to control and monitor the spacecraft systems, and equipment required for the comfort and safety of the crew. The primary structure of the command module is encompassed by three heat shields, forming a conical-shape exterior. The forward, crew, and aft heat shield structures are coated with ablative material and joined to the primary structure. An insulation material is installed between the primary structure and the heat shields. The C/M consists of three compartments: forward, crew, and aft.

2-10. FORWARD COMPARTMENT. The forward compartment (figure 2-4) is a section between the forward heat shield and the forward side of the forward pressure bulkhead. The center portion is occupied by a forward tunnel which permits crew members to transfer to the LEM and return to the crew compartment during the performance of lunar mission tasks. The interior of the forward compartment is divided into four 90-degree segments which contain earth landing system (ELS) components, recovery equipment, two reaction control motors, and the heat shield jettisoning mechanism. The major portion of this section contains the active components of the ELS consisting of three main parachutes, three pilot parachutes, two drogue parachutes, as well as drogue and pilot parachute

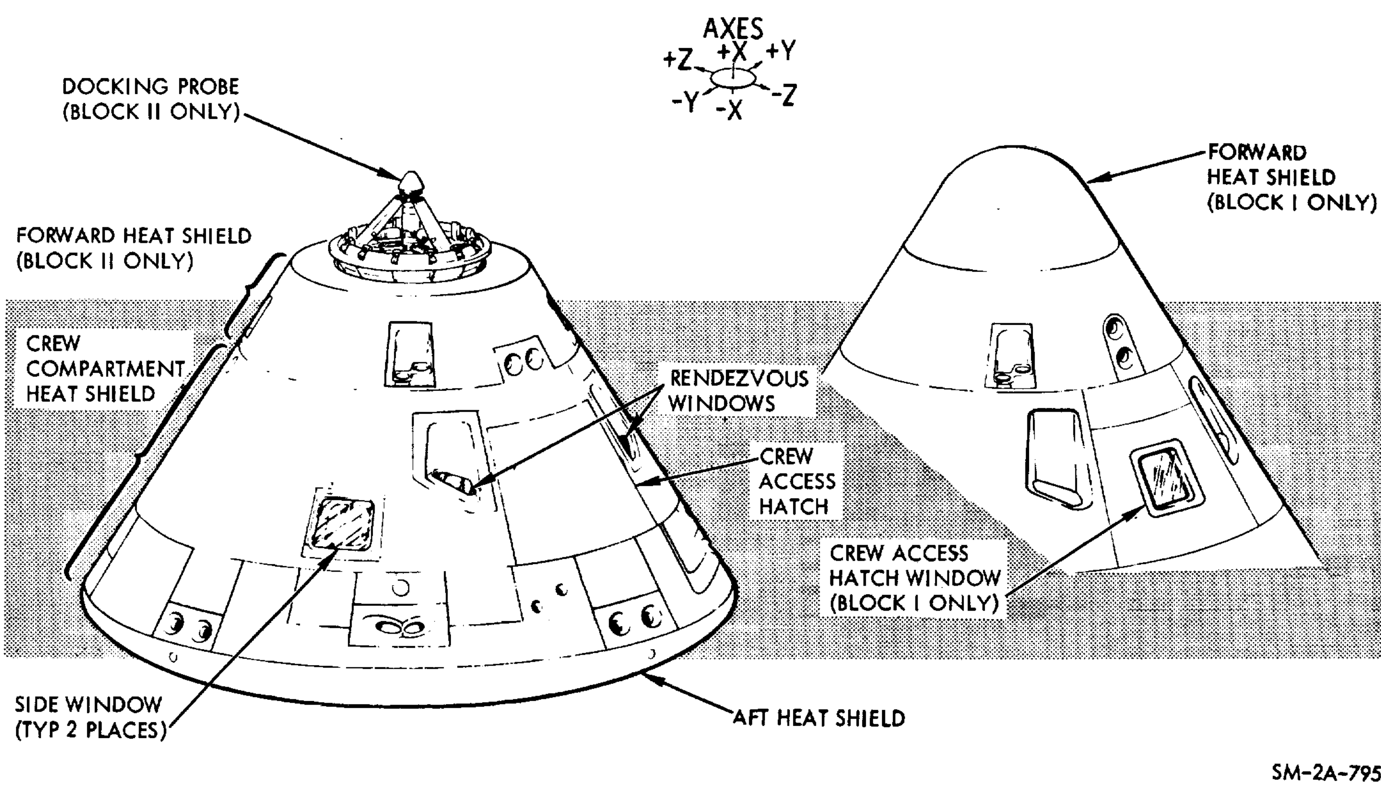


Figure 2-3. Command Module

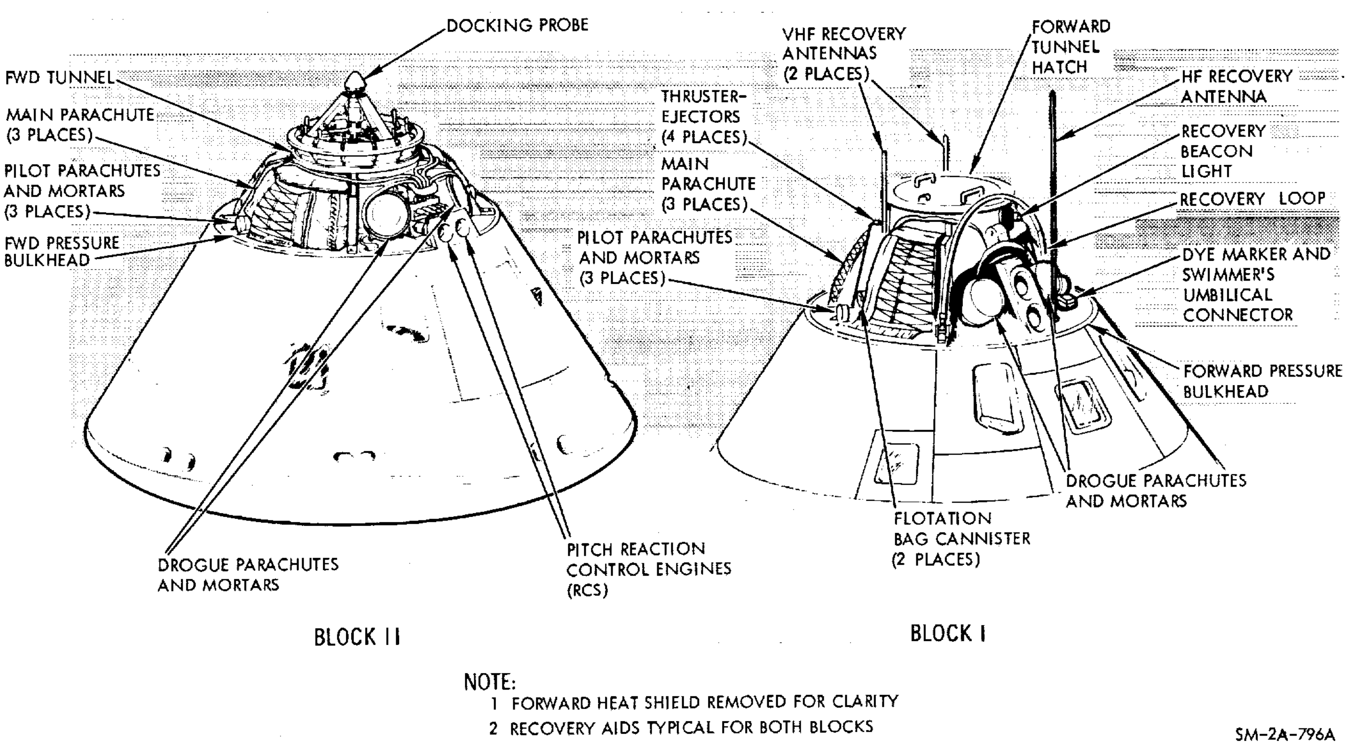


Figure 2-4. Command Module Forward Compartment

mortars, risers, and the necessary hardware. The recovery equipment, installed in the forward compartment, consists of three flotation bags of the uprighting system, a beacon light, a sea dye marker, a swimmer umbilical, three recovery antennas and a recovery pickup loop. Four thruster-ejectors are installed in the forward compartment to eject the forward heat shield during landing operations. The thrusters operate in conjunction with the heat shield release mechanism to produce a rapid, positive release of the heat shield, preventing parachute fabric damage.

2-11. CREW COMPARTMENT. The crew compartment (figure 2-5) is a sealed, three-man cabin with pressurization maintained by the environmental control system. The crew compartment contains the necessary systems, spacecraft controls and displays, observation windows, access hatches, food, water, sanitation, and survival equipment. The compartment incorporates windows and equipment bays as a part of the structure. The following listing contains specific items contained in the crew compartment and their locations. Items marked (Block I) or (Block II), indicate the specific block in which they are included; all others are included in both blocks.

Aft Equipment Storage Bay (See figure 2-5.)

Space suits (two)

Space suit spare parts

Umbilicals

Rest station restraint

Drogue and probe stowage (Block II)

Life vests (three)

TV zoom lens

Portable life support system (Block II)

CO_2-odor absorber filters

Fecal canister

Helmet storage (two)

Communication helmet storage (Block I)

Lower Equipment Bay (See figure 2-5.)

Power servo assembly (PSA)

Computer control panel

Signal conditioning equipment

Rate gyro assembly (RGA) (Block II)

Electronic control amplifiers (five) (Block I)

Gas chromatograph (Block I)

Workshelf (Block I)

Pulse-code modulation (PCM) units (two)

S-band power amplifier

Unified S-band equipment and spares

Junction box

Motor switches (three)

Battery charger

Sextant and telescope

Rendezvous radar control (Block II)

TVC servo amplifier (Block II)

Control electronics (Block II)

Gyro display (Block II)

Display electronics (Block II)

Reaction jet driver (Block II)

Apollo guidance computer (AGC)

Medical supplies

Medical refrigerator (Block II)

Data distribution panel (Block I)

Attitude gyro-accelerometer assembly (AGAA)

Crew flight data file (CFDF)

VHF multiplexer (Block I)

C-band transponder (Block I)

Audio center equipment

Central timing equipment (CTE)

Entry batteries (three)

Circuit breaker panel

Guidance and navigation (G&N) control panel

Coupling display unit (CDU) (Block II)

Data storage equipment

Food storage

Scientific equipment

Flight qualification recorder (Block I)

Up-data link

Premodulation processor

VHF/AM transceiver and VHF recovery beacon

Triplexer (Block II)

VHF/FM transmitter and HF transceiver

R-F switch

Inverters (three)

A-C power box

Pyrotechnic batteries (two)

Lighting control (Block II)

Clock and event timer (Block II)

In-flight tool set

Left-Hand Forward Equipment Bay (See figure 2-5.)

Cabin air fan

Optical storage (Block I)

Translation controller connector panel

Loose parts storage (Block I)

Pressure suit connectors (three)

Water delivery assembly

Food reconstitution device

Clothing (storage)

Cabin heat exchanger

Clock and event-timer panel (Block I)

Thermal radiation coverall (storage) (Block I)

Radiation survey meter (Block II)

Left-Hand Equipment Bay (See figure 2-5.)

Cabin pressure relief valve

Fixed shock attenuation panel

Environmental control system (ECS) water and oxygen control panels

Surge tank

Environmental control unit

Pressure hatch stowage (Block II)

CO_2 sensor

Removable shock attenuation panels (two)

Right-Hand Forward Equipment Bay (See figure 2-5.)

Individual survival kits (three on Block I S/C) (one on Block II S/C)

System test equipment (Block I)

Waste storage inlet

TV camera and mount food (Block I)

Optical storage (Block II)

Medical supplies

LEM docking target storage (Block II)

Bio-instrument accessories (Block II)

Sanitary supplies

Tools and belt (Block II)

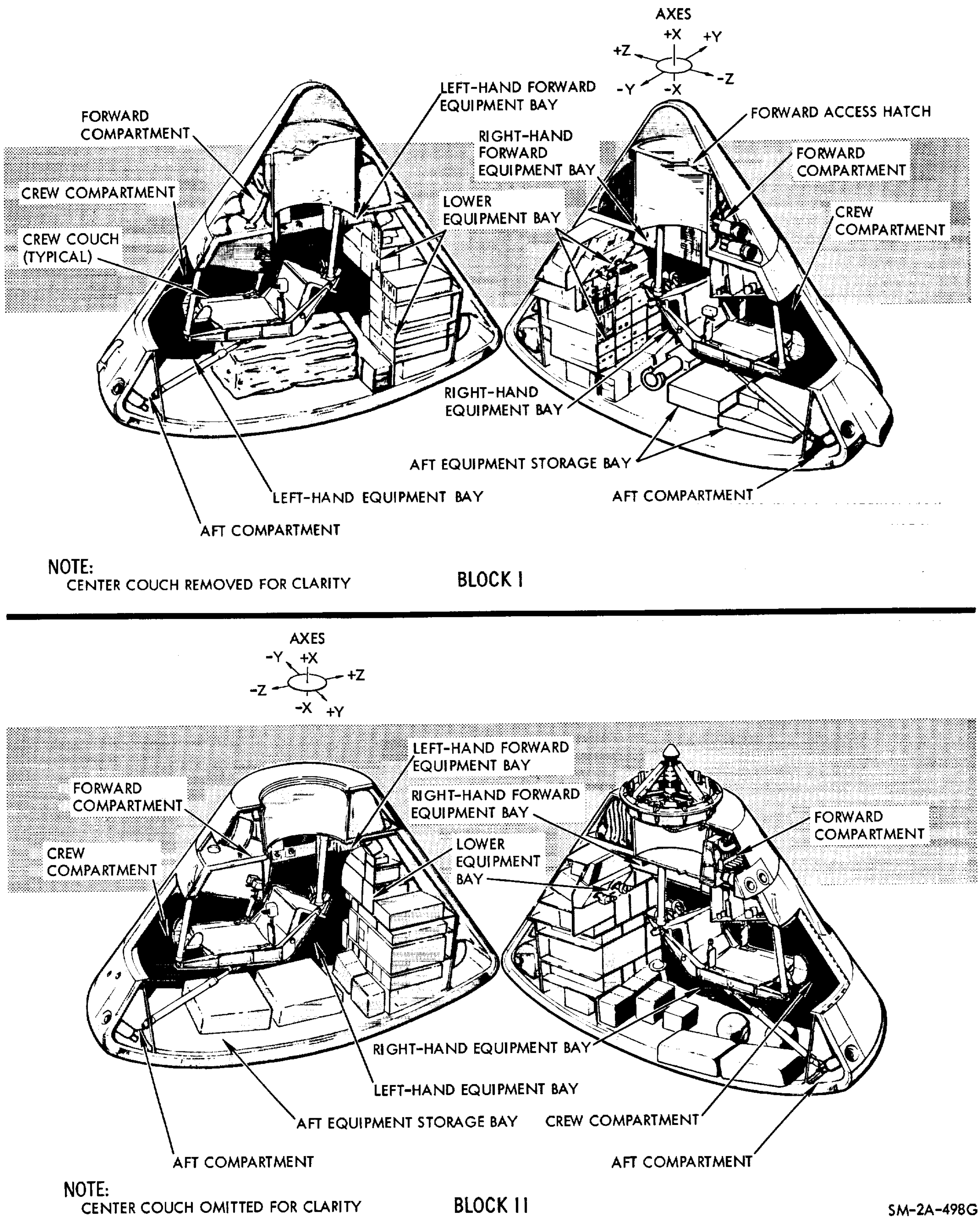


Figure 2-5. Command Module Compartments and Equipment Bays (Typical)

Right-Hand Equipment Bay (See figure 2-5.)

Vacuum cleaner	Fuse box
Electrical power equipment	Food (Block II)
Mission sequencer	Waste storage compartment
Power distribution box	ELS sequencer
Circuit utilization box	Mission sequencer
Phase correction capacitor box	Signal conditioners
Waste management system controls	Thermal hatch stowage (Block II)

2-12. AFT COMPARTMENT. The aft compartment (figure 2-5) is an area encompassed by the aft portion of the crew compartment heat shield, aft heat shield, and aft sidewall of the primary structure. This compartment contains 10 reaction control motors, impact attenuation structure, instrumentation, and storage tanks for water, fuel, oxidizer, and gaseous helium.

2-13. SERVICE MODULE.

2-14. The service module structure is a cylinder formed by six panels of one-inch aluminum honeycomb. (See figure 2-6.) Its interior is unsymmetrically divided into six sectors by radial beams, or webs fabricated of milled aluminum alloy plate. The interior consists of diametrically opposed sectors around a circular section 44 inches in diameter. Sectors 1 and 4 are 50-degree segments and house essential auxiliary equipment. Sectors 2 and 5 are 70-degree segments each containing a large oxidizer tank for the service propulsion system. The remaining sectors, 3 and 6, are 60-degree segments and hold fuel tanks for the service propulsion system. The equipment contained within the S/M is accessible through maintenance doors located strategically around the exterior surface of the module. The specific items contained in the S/M compartments, and their location, are listed in paragraph 2-16.

2-15. An area between the service and command modules provides space for radial beam trusses connecting these two modules. Beams one, three, and five have compression pads for support of the C/M; beams two, four, and six have compression pads, shear pads, and tension ties. A flat center section in each tension tie, incorporates redundant explosive charges for service module-command module separation. The entire separation system is enclosed within a fairing 26 inches high and 13 feet in diameter.

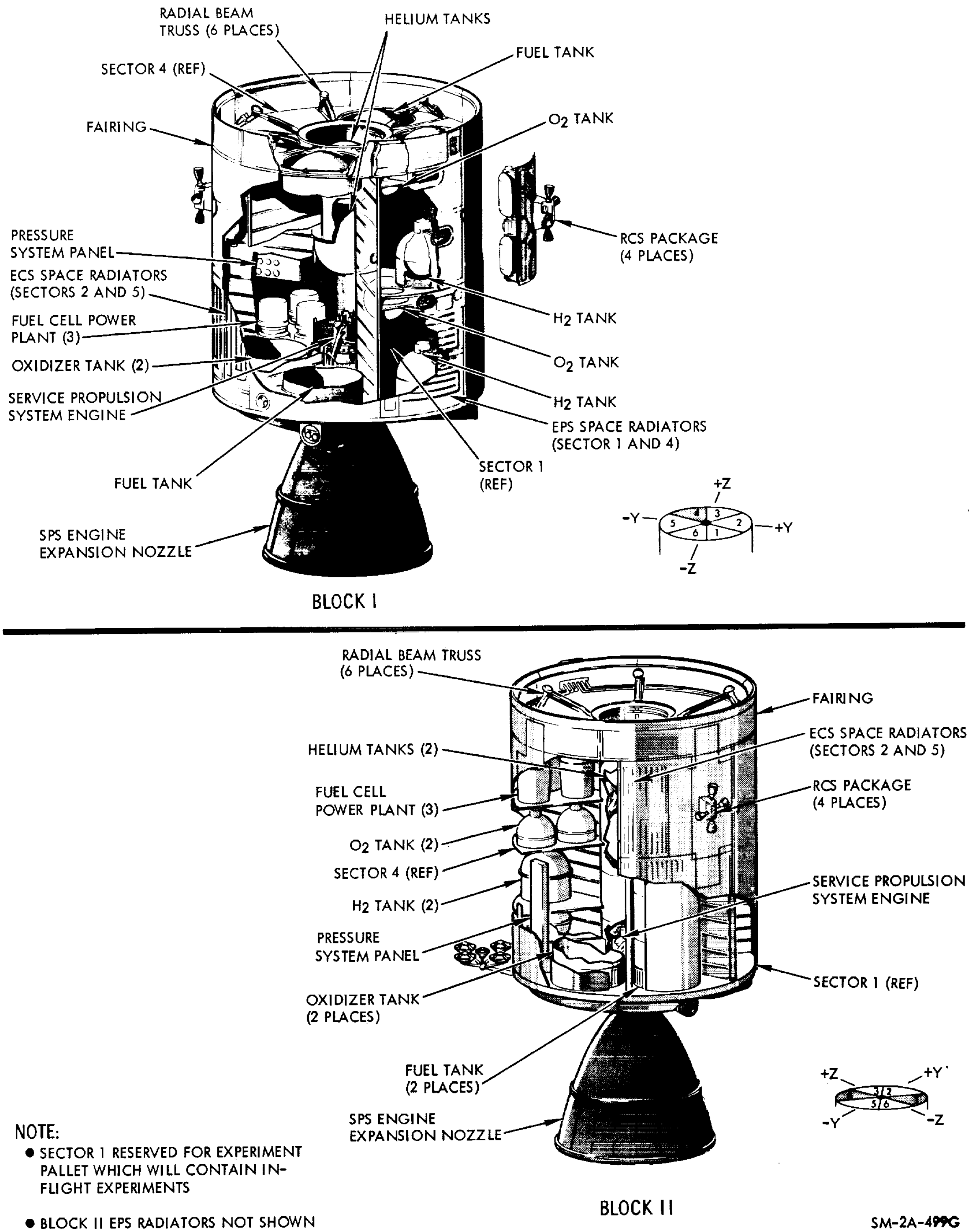


Figure 2-6. Service Module

2-16. Items and their locations contained in Block I and Block II service modules (manned S/C only) are listed as follows:

Location	Contents: Block I	Contents: Block II
Sector 1	Electrical power system space radiators Super-critical oxygen tank (two) Super-critical hydrogen tank (two)	Experiment pallet-reserved for in-flight experiments
Sector 2	Environmental control system space radiator Service propulsion system oxidizer tank Reaction control system engine cluster (+Y-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank Space radiator isolation valve (two)	Environmental control system space radiator Service propulsion system oxidizer tank Reaction control system engine cluster (+Y-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank Space radiator isolation valve (two)
Sector 3	Service propulsion system fuel tank Reaction control system engine (cluster (+Z-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank Rendezvous radar transponder	Service propulsion system fuel tank Reaction control system engine cluster (+Z-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank Rendezvous radar transponder
Sector 4	Electrical power system space radiator Fuel cell power plant (three) Helium distribution system Reaction control system control unit Electrical power system power control relay box High-gain antenna (stowed under) Service module jettison controller (SMJC) sequencer (two)	Electrical power system space radiator Fuel cell power plant (three) Super-critical oxygen tank (two) Super-critical hydrogen tank (two) Reaction control system control unit Electrical power system power control relay box High-gain antenna (stowed under) Service module jettison controller (SMJC) sequencer (two)

Location	Contents	
	Block I	Block II
Sector 5	Environmental control system space radiator Service propulsion system oxidizer tank Reaction control system engine cluster (-Y-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank	Environmental control system space radiator Service propulsion system oxidizer tank Reaction control system engine cluster (-Y-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank Helium distribution system
Sector 6	Space radiator selection valve (two) Glycol shutoff valve (two) Reaction control system engine cluster (-Z-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank Service propulsion system fuel tank	Space radiator selection valve (two) Glycol shutoff valve (two) Reaction control system engine cluster (-Z-axis) Reaction control system helium tank Reaction control system fuel tank Reaction control system oxidizer tank Service propulsion system fuel tank
Center section	Service propulsion system helium tank (two) Service propulsion system engine	Service propulsion system helium tank (two) Service propulsion system engine

2-17. LUNAR EXCURSION MODULE.

2-18. The LEM, manufactured by Grumman Aircraft Engineering Corp., is a space vehicle which provides a means of transportation in a space environment for two crew members of the Apollo spacecraft. During a lunar landing mission, the LEM is required to enable two astronauts to leave the command module, land on the lunar surface, and return to the command module. The LEM is then jettisoned from the C/M and left as a lunar satellite. A description of the LEM is presented in section IV.

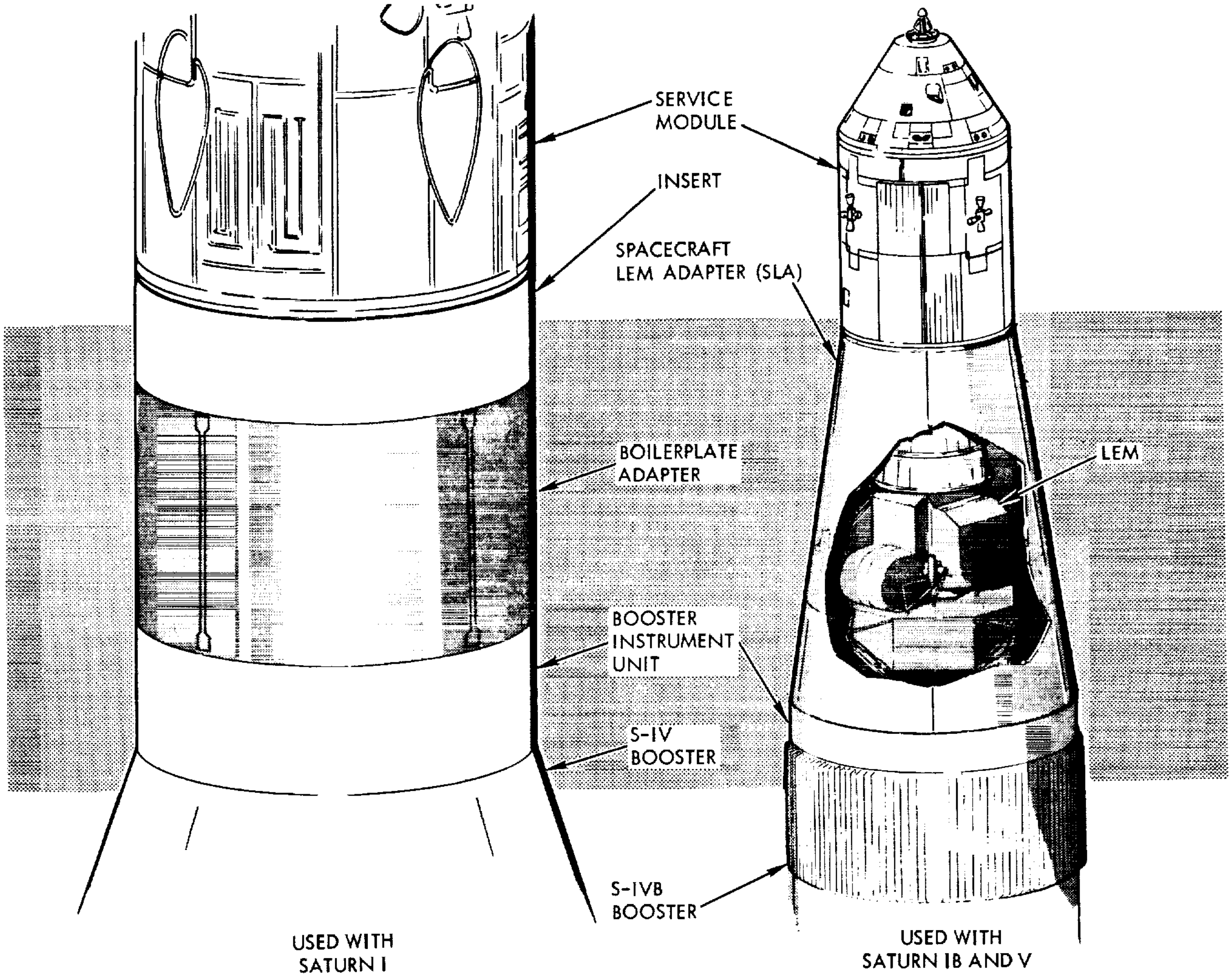


Figure 2-7. Spacecraft Adapters

2-19. BOILERPLATE AND SPACECRAFT LEM ADAPTERS.

2-20. The boilerplate and spacecraft LEM adapters (figure 2-7) consist of the structural interstage between the launch vehicle and the spacecraft. The boilerplate adapter is required on test vehicles using a Saturn I launch vehicle. (See figure 2-8.) The spacecraft LEM adapter (SLA) is required on Apollo spacecraft using a Saturn IB or Saturn V launch vehicle. (See figure 2-8.) The SLA will house the service propulsion engine expansion nozzle, high-gain antenna, and LEM.

2-21. The different adapters used during the missions to mate the spacecraft to the launch vehicle are necessitated by the varying sizes of the launch vehicles. In each instance, an umbilical cable is incorporated in the adapter to connect circuits between the launch vehicle and the spacecraft.

2-22. The SLA (figure 2-7) is a tapered cylinder comprised of eight panels, four of which have linear explosive charges installed at panel junctions. During CSM/SLA separation, the charges are fired to, open the four panels, free the spacecraft from the launch vehicle, and expose the LEM.

2-23. LAUNCH VEHICLES.

2-24. Launch vehicles used in the Apollo program are illustrated in figure 2-8. The earlier test evaluation and qualification flight vehicles are powered by the launch escape vehicle, Little Joe II, Saturn I, and Saturn IB launch vehicles. As the Apollo program progresses, the extended lunar mission performance and greater payload necessitates the use of the Saturn V. The general configurations of the launch vehicle boosters are summarized in paragraphs 2-25 through 2-33.

2-25. LAUNCH ESCAPE VEHICLE.

2-26. The launch vehicle used for pad-abort tests (figure 2-2) consists of a C/M, boost protective cover, launch escape tower, launch escape system motor, a pitch control motor, and a tower jettison motor. Each of the motors uses a solid propellant. The LES motor, manufactured by Lockheed Aircraft Corporation, provides up to 150, 000 pounds of thrust. The pitch control motor, also manufactured by Lockheed Aircraft Corporation, provides up to 3, 400 pounds of thrust. The tower jettison motor, manufactured by Thiokol Chemical Corporation, provides up to 33, 000 pounds of thrust.

2-27. LITTLE JOE II.

2-28. The Apollo transonic and high-altitude abort tests will utilize the Little Joe II launch vehicle. (See figure 2-8.) The launch vehicle is approximately 13 feet in diameter and 29 feet in length. Little Joe II, manufactured by General Dynamics, Convair, uses a combination of Algol and Recruit solid-propellant motors. One Algol and six Recruit motors provide an initial boost of 310, 000 pounds of thrust.

2-29. SATURN I.

2-30. Saturn I consists of an S-I first-stage booster and an S-IV second stage. (See figure 2-8.) The S-I, manufactured by Chrysler Corporation, is 257 inches in diameter and approximately 82 feet in length. Eight Rocketdyne H-I engines are used, each burning RP-1 and liquid oxygen, and each producing 200, 000 pounds of thrust. Total boost for the S-I is 1, 500, 000 pounds of thrust. The S-IV, manufactured by Douglas Aircraft Company, is 220 inches in diameter and 40 feet in length. Six Pratt & Whitney RL-10 engines are used, each burning liquid hydrogen and liquid oxygen and each producing 15, 000 pounds of thrust. The total boost for the S-IV is 90, 000 pounds. An instrument unit, located between the S-IV and the boilerplate adapter, controls each of the two stages during flight.

2-31. SATURN IB.

2-32. Saturn IB is a more powerful version of Saturn I, consisting of an S-IB first-stage booster and an S-IVB second stage. (See figure 2-8.) The S-IB, manufactured by Chrysler Corporation, is a lightweight version of the S-I booster, but approximately the same size. A weight reduction of 15, 000 pounds per engine, or 120, 000 pounds total, is realized while maintaining 1, 600, 000 pounds of thrust. The S-IVB, manufactured by Douglas Aircraft Company, is 260 inches in diameter, 58 feet in length, and of an entirely different configuration than the S-IV. S-IVB employs a single Rocketdyne J-2 engine, burning liquid hydrogen and liquid oxygen, to produce approximately 200, 000 pounds of thrust. An instrument unit, located between the S-IVB and the SLA, controls each of the two stages during flight.

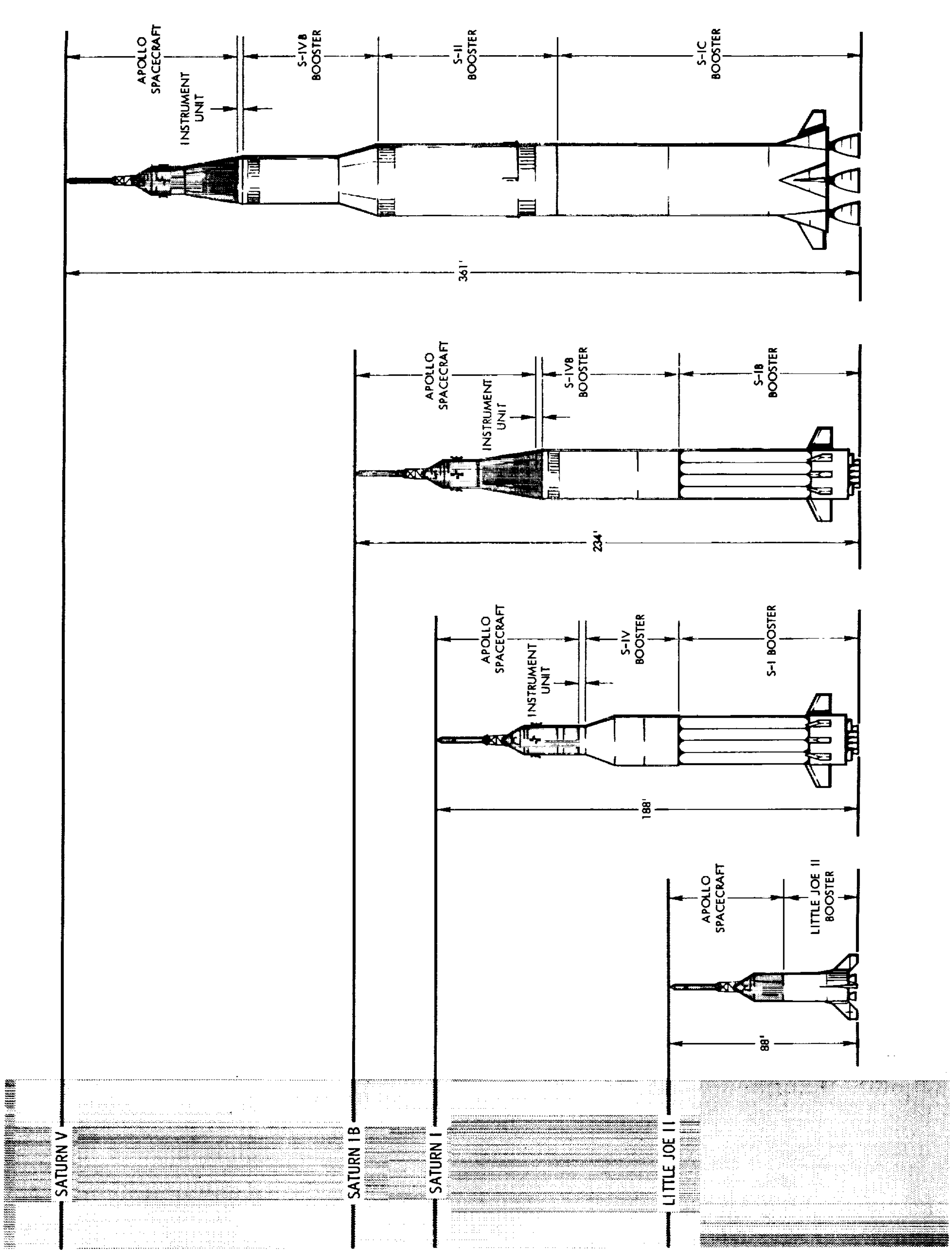


Figure 2-8. Launch Vehicle Configurations

2-33. SATURN V.

2-34. Saturn V is a three-stage launch vehicle consisting of an S-IC first-stage booster, S-II second stage, and an S-IVB third stage. (See figure 2-8.) The S-IC, manufactured by the Boeing Company, is 33 feet in diameter and 138.5 feet in length; it uses five Rocketdyne F-1 engines. Each F-1 engine, burning RP-1 and liquid oxygen, produces 1,500,000 pounds of thrust for an overall boost of 7,500,000 pounds. The S-II, manufactured by the Space and Information Systems Division of North American Aviation, Inc., is 33 feet in diameter and approximately 82 feet in length and employs five Rocketdyne J-2 engines. Each J-2 engine, burning liquid hydrogen and liquid oxygen, produces 200,000 pounds of thrust for an overall boost of 1,000,000 pounds. The S-IVB is similar to the second stage of Saturn IB, producing 200,000 pounds of thrust. An instrument unit, located between the S-IVB and the SLA, controls each of the three stages during flight.

SPACECRAFT SYSTEMS

A0118

3-1. GENERAL.

3-2. This section contains data relative to the basic nature of the operational command and service module spacecraft systems. The purpose of each system, its functional description, and interface information are presented with a minimum of detailed text consistent with understanding. Concepts are supported by illustrations, listings, and diagrams. Also illustrated are the various panel arrangements within the command module that contain the controls and displays. Data presented covers each complete system as it will be installed in a manned spacecraft for an earth orbital or a lunar landing mission. Systems and components will be tested and qualified prior to manned missions. Other missions, using boilerplates or unmanned S/C, are not covered in this section as these vehicles may contain incomplete or modified systems. Physical differences between Block I and Block II S/C are illustrated in this section. For an explanation of Block I and Block II, refer to section VII.

3-3. Redundancy is necessary for mission critical items throughout the spacecraft systems in order to maintain the high reliability rates prescribed for the Apollo program. Included are redundant components, power sources, paths for fluids and electrical signals, and redundant operational procedures.

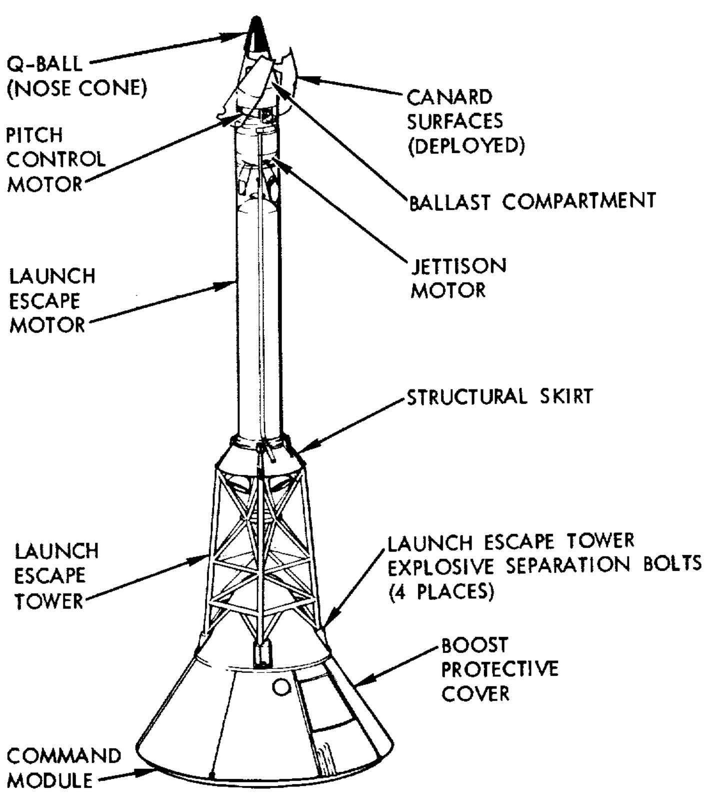
3-4. LAUNCH ESCAPE SYSTEM.

3-5. The launch escape system (LES) provides immediate abort capabilities from the launch pad, or away from the path of the launch vehicle in the event of a pad abort or an abort shortly after launch. (See figure 3-1.) Upon abort initiation, the command module will be propelled to a sufficient altitude and lateral distance away from the danger area for the effective operation of the earth landing system. Upon completion of an abort, or a successful launch, the launch escape assembly is jettisioned from the C/M utilizing the tower jettison motor.

3-6. The LES consists of two major structures, plus electrical control and regulating equipment located in the C/M. The forward structure is cylindrical, housing three rocket motors and a ballast compartment topped by a Q-ball (nose cone). Two canard surfaces are installed below the nose cone. The rocket motors are loaded with solid propellants of various grain patterns, depending upon motor performance requirements. The aft structure is a four-leg, welded, tubular titanium tower. It serves as an intermediate structure, transmitting loads between the C/M and the launch escape assembly, and positioning the C/M a suitable distance from the launch escape motor exhaust. (See figure 3-1.) At its forward end, the tower is attached to a structural skirt that covers the exhaust nozzles of the launch escape motor, and at the aft end, by four explosive bolts attached to the C/M. A boost protective cover is also attached to the aft end of the tower to protect the C/M from the launch escape motor exhaust and boost heating. The LES sequence controllers are located in the C/M and control the system by transmitting signals that ignite the rocket motors, deploy the canard surfaces, and detonate the separation devices. The Q-ball has four static ports for measuring ΔP which is a function of angle of attack. An angle-of-attack indicator is located on the main display console and displays the combined pitch and yaw vectors in terms of percentages.

3-7. LES OPERATION.

3-8. The LES is initiated automatically by the emergency detection system (EDS) of the launch-boost vehicle, or manually by the astronauts at any time from pad to launch escape assembly jettison altitude, as shown in figure 3-3. Upon receipt of an abort signal, regardless of its source, the booster is cut off (after the first 40 seconds of flight), the CSM separation device is detonated, and, subsequently, the simultaneous ignition of the launch escape and pitch control motors takes place. These motors provide sufficient thrust for the lift and lateral translation of the C/M away from the launch pad or trajectory of the launch vehicle where, at a safe altitude, the launch escape assembly is jettisoned



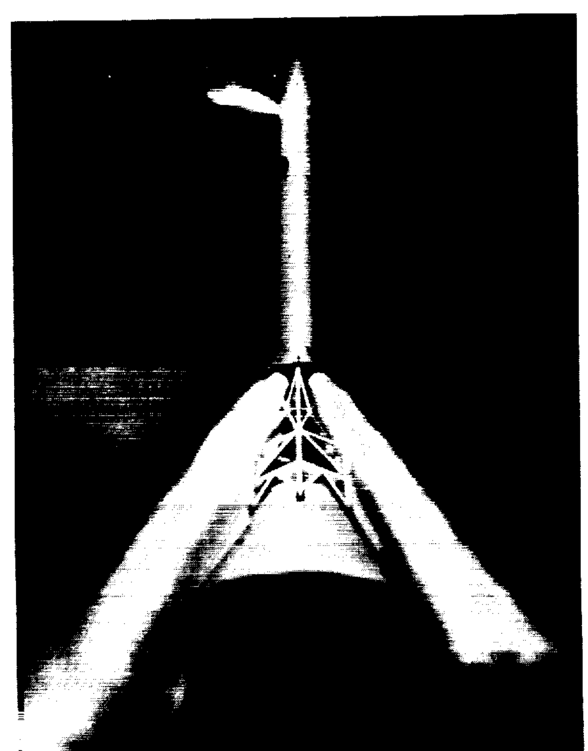


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Figure 3-1. Launch Escape Vehicle

and the earth landing system activated. Approximately 11 seconds after abort initiation, the canard surfaces are deployed; 3 seconds later, or at approximately 24,000 feet, the tower explosive bolts are detonated and the tower jettison motor is ignited. These actions will carry the spent launch escape assembly and boost protective cover away from the C/M trajectory and impact zone; 0.4 second later the apex cover (forward heat shield) is jettisioned, and the earth landing system (ELS) is activated. The canard surfaces are not deployed for an abort above 120,000 feet altitude since the LES tower is manually jettisoned between 6 to 10 seconds after abort initiation.

3-9. During a successful launch, the launch escape assembly will be manually jettisoned after reaching a prescribed altitude. The tower explosive bolts will be detonated, the jettison motor ignited, and the launch escape assembly (including the boost protective cover) will be propelled away from the path of the spacecraft and booster.

3-10. CANARD OPERATION. The canard (figure 3-2) consists of two deployable surfaces and an operating mechanism. The surfaces are faired into the outer skin of the launch escape assembly below the nose cone and the operating mechanism is inside the cylindrical-shaped assembly. Each surface is mounted on two hinges and is opened by a pyro cylinder

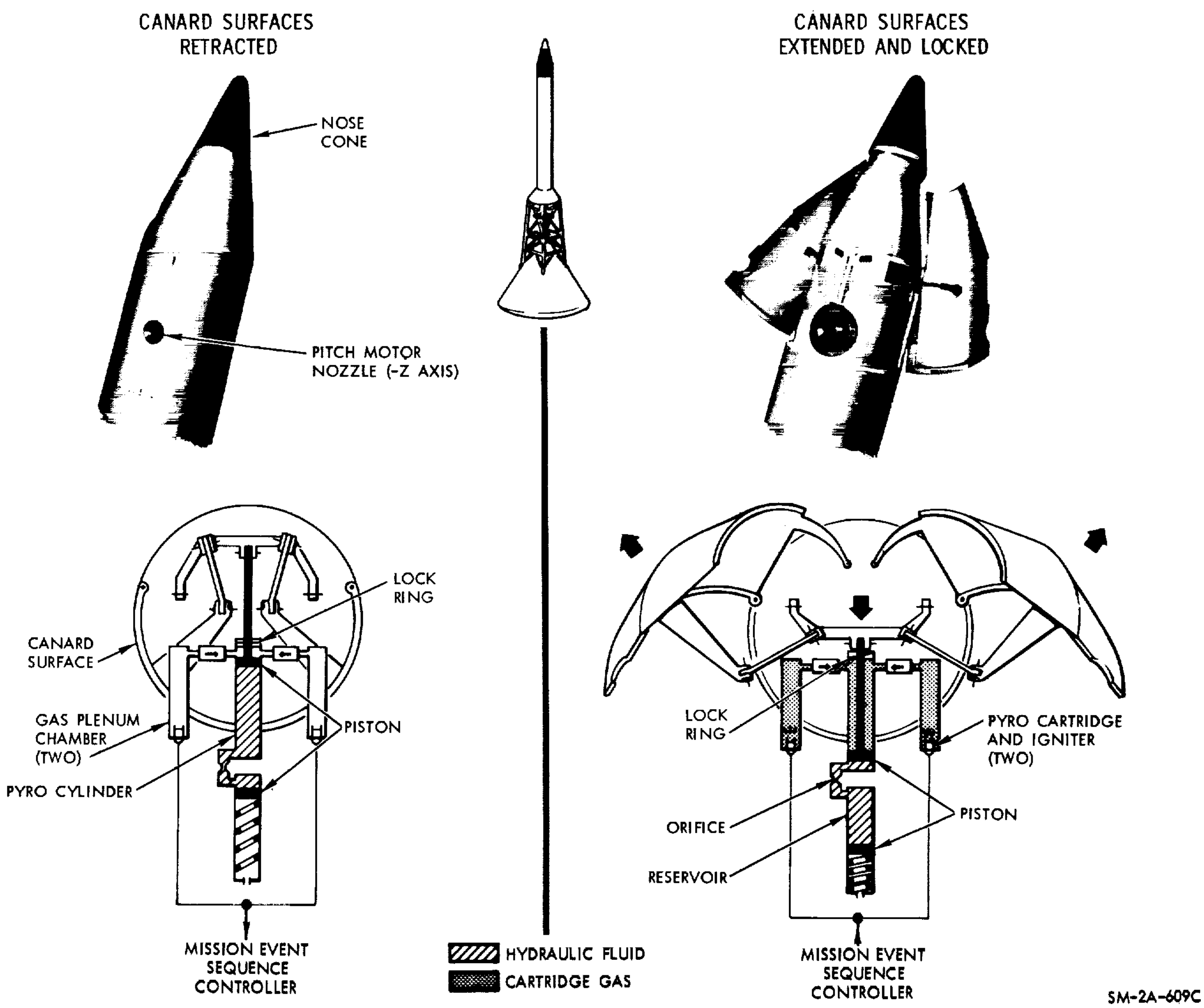


Figure 3-2. Canard Operation

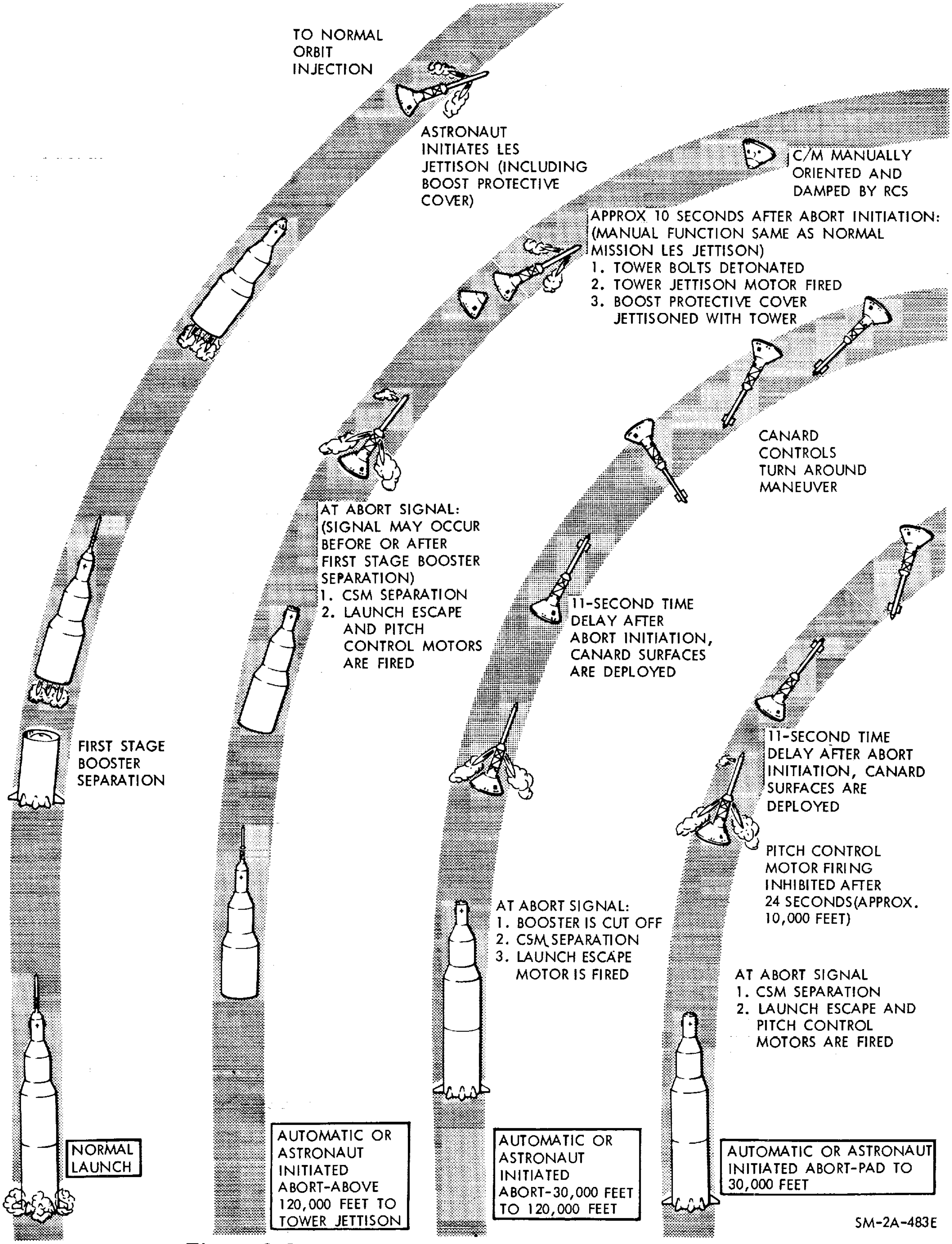


Figure 3-3. Launch Escape and Earth Landing Systems Functional Diagram (Sheet 1 of 2)

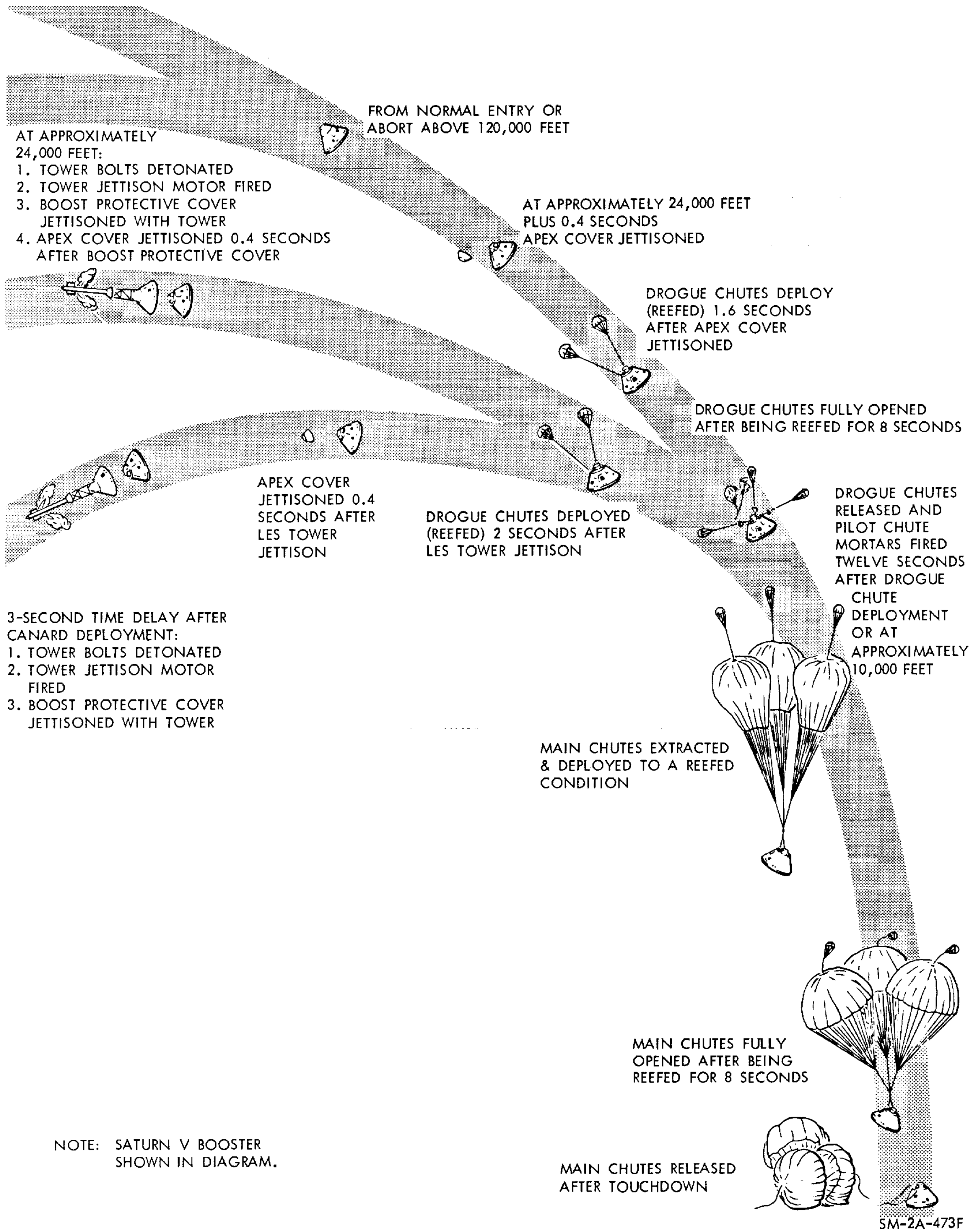


Figure 3-3. Launch Escape and Earth Landing Systems Functional Diagram (Sheet 2 of 2)

that operates the opening mechanism. The pyro cylinder piston is normally in the extended position with the canard surfaces closed. Eleven seconds after an abort signal is received by the launch escape system, an electrical current fires two pyro cartridges to open the canard surfaces. Gas from the cartridges causes the piston to retract, operating the opening mechanism. The cylinder is filled with hydraulic fluid downstream of the piston and, as the piston retracts, the fluid is forced out through an orifice into a reservoir. Metering the fluid through the orifice controls the speed of the canard operation. When fully open, the two canard surfaces are locked in place by gas pressure in the cylinder, a lock ring on the piston shaft, and an overcenter linkage. The induced aerodynamic forces acting on the canard surfaces will orient the C/M to a blunt-end-forward trajectory before launch escape assembly jettison and ELS activation. (See figure 3-3.)

3-11. EMERGENCY DETECTION SYSTEM.

3-12. The emergency detection system (EDS) is designed to detect and display emergency conditions of the launch vehicle to the astronaut. The EDS also provides automatic abort initiation, under certain conditions, between lift-off and LES tower jettison. The display circuitry and automatic abort capabilities are enabled at lift-off. A lockout system is provided to prevent enabling the automatic abort circuitry prior to lift-off. The astronaut may initiate an abort manually at any time with the commanders translation control.

3-13. AUTOMATIC ABORT.

3-14. The emergency detection system (EDS) will initiate an automatic abort signal after launch by sensing excessive vehicle rates or two engines out. The abort signal will cause booster cutoff, event timer reset, and launch escape system activation.

3-15. MANUAL ABORT.

3-16. A manual abort (figure 3-4) can be initiated prior to, or during launch by manual CCW rotation of the commanders translation control. The launch escape system will be utilized until approximately 168 seconds after lift-off. During a normal mission, the LES tower is jettisoned shortly after second booster stage ignition, and any abort thereafter is accomplished by utilizing the SPS engine in the service module. An SPS abort must be manually initiated. Upon abort initiation, the booster automatically separates from the spacecraft, S/M-RCS engines fire to accomplish the ullage maneuver, and the SPS engine ignites to thrust the S/C away from the booster.

3-17. ABORT REQUEST INDICATOR LIGHT AND EVENT TIMER.

3-18. The ABORT request indicator light is illuminated by ground control, or the range safety officer, using GSE or a radio command through the up-data link. When illuminated, the light indicates an abort request and serves to alert the crew of an emergency situation.

3-19. Initiation of an abort (automatic or manual) will automatically reset the event timer to provide a time reference for manual operations.

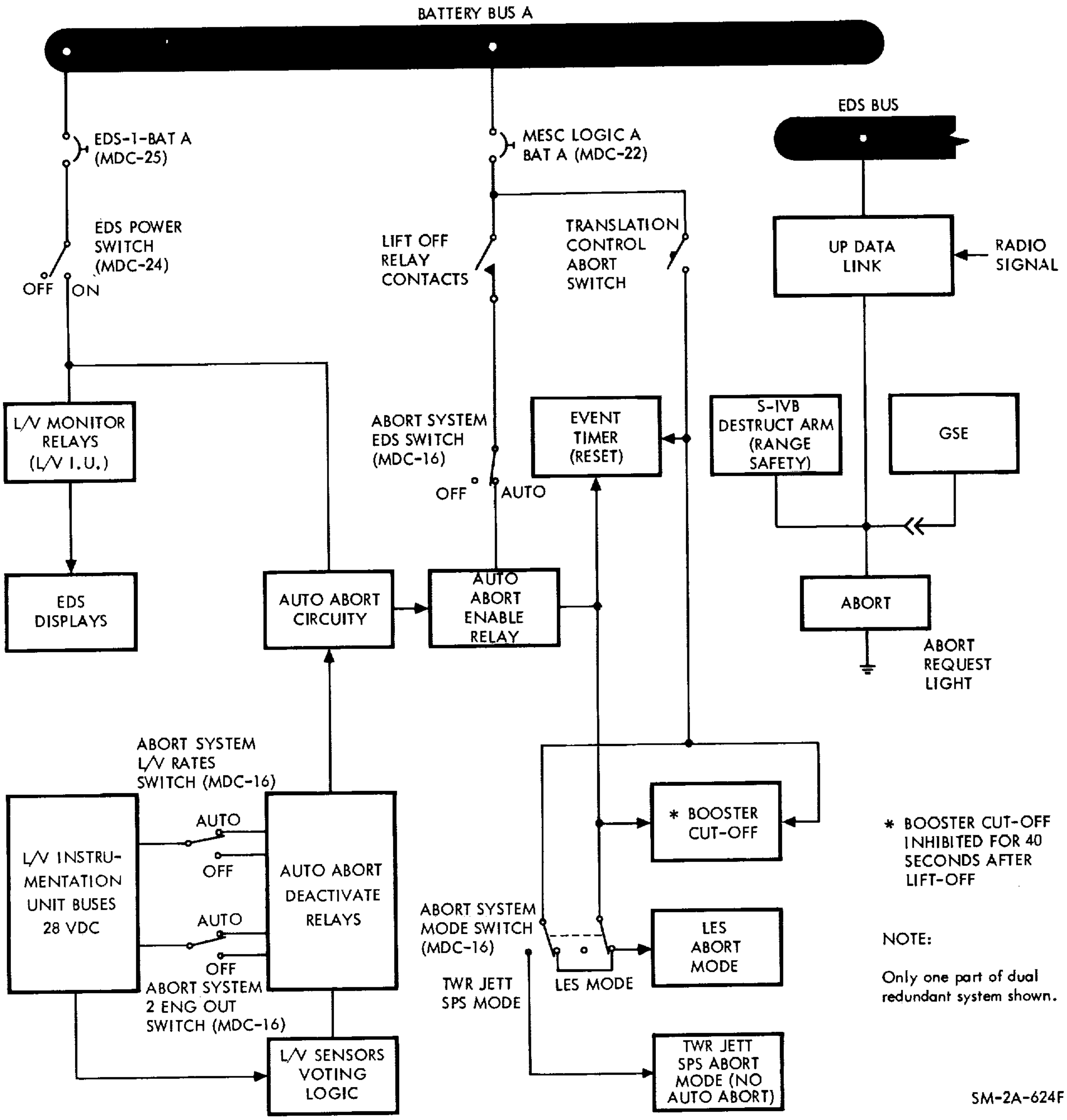


Figure 3-4. Emergency Detection System Automatic and Manual Abort Block Diagram

3-20. EARTH LANDING SYSTEM.

3-21. The purpose of the earth landing system (ELS) is to provide a safe landing for the astronauts and the command module following an abort, or a normal entry from an earth orbital or lunar landing mission. (See figure 3-3.) Included as a part of the ELS are several recovery aids which are activated after impact on either land or water. The ELS operation is automatically timed and activated by sequence controllers. There are, however, backup components and manual override controls to ensure system reliability and to provide astronaut control.

3-22. With the exception of the controls in the C/M crew compartment, ELS components are located in the forward compartment of the C/M as shown in figure 3-5. The ELS consists of the forward heat shield ejection subsystem, the sequence controllers, recovery aids, and the parachute subsystem. The parachute subsystem, is comprised of two fist ribbon nylon drogue parachutes, 13.7 feet in diameter; three ring slot nylon pilot parachutes, 7.2 feet in diameter; three ring sail nylon main parachutes, 83.5 feet in diameter; deployment bags, harness, mortars, and the necessary hardware for attachment to the C/M.

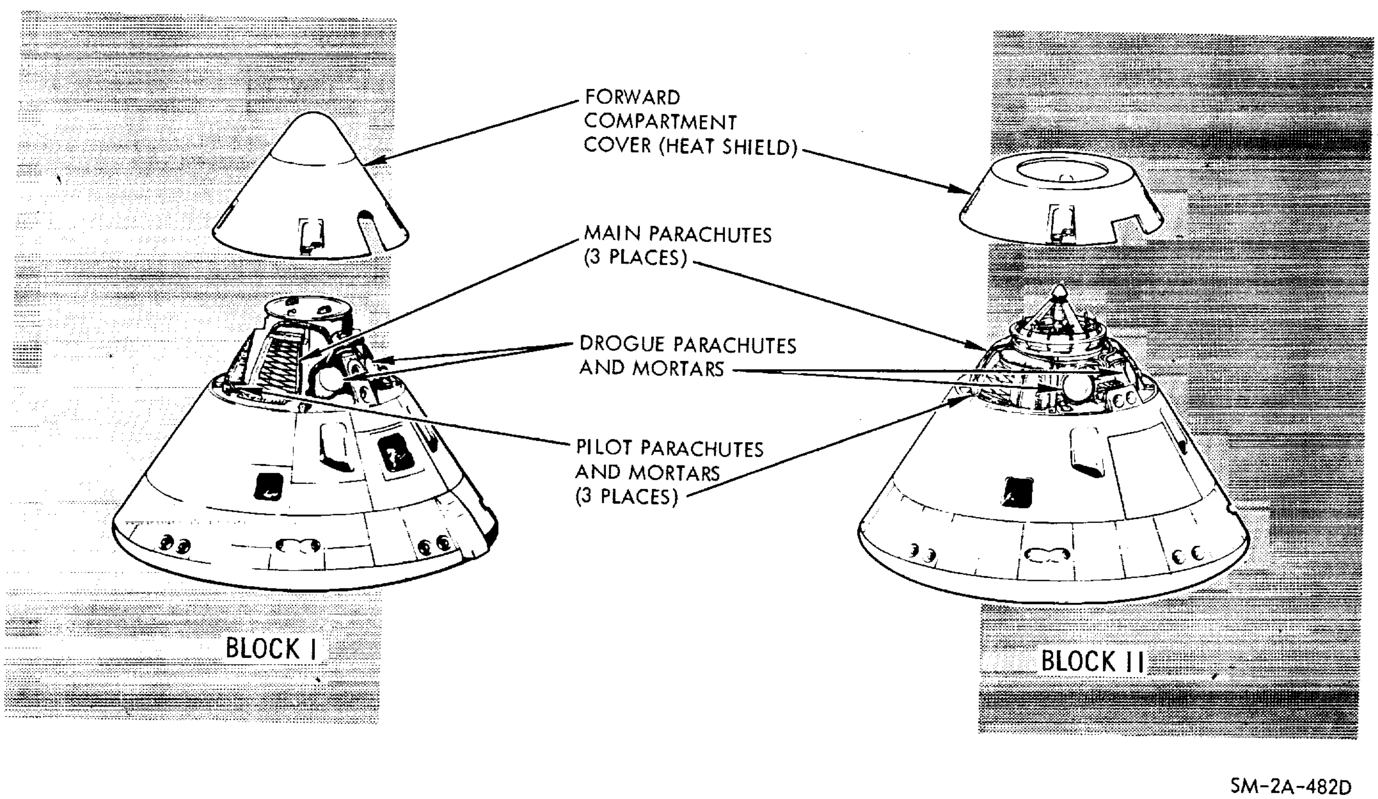


Figure 3-5. Earth Landing System

3-23. ELS OPERATION.

3-24. The C/M-ELS begins operation upon descending to approximately 24,000 feet, or, in the event of an abort, after launch escape assembly jettison. (See figure 3-3.) The apex cover (forward heat shield) is jettisoned by four gas-pressure thrusters, or, in the event of a pad or low-altitude abort, is jettisoned 0.4 second after the launch escape tower is jettisoned. This function is imperative, as the forward heat shield covers and protects the ELS parachutes up to this time. At 1.6 seconds later, the drogue mortar pyrotechnic cartridges are fired to deploy two drogue parachutes in a reefed condition. After 8 seconds, the reefing lines are severed by reefing line cutters and the drogue parachutes open. These orient the C/M in a blunt-end-forward attitude and provide deceleration. The drogue parachutes are then released, and the three pilot parachute mortars are simultaneously fired. This action ejects the pilot parachutes which, in turn, extract and deploy the three main parachutes. To preclude the possibility of main parachute damage or failure due to excessive descent velocity, the main parachutes open to a reefed condition for 8 seconds to further decelerate the C/M. The three parachutes are then fully opened (disreefed) to lower the C/M at a predetermined descent rate. A 27-1/2-degree hang angle of the C/M is achieved by the main parachute attachment points. In the event one main parachute fails to open, any two parachutes will safely carry out the prescribed function. The main parachutes are disconnected following impact, and the recovery aid subsystem is set in operation by the crew. The recovery subsystem consists of an uprighting system, swimmers umbilical, and sea (dye) marker for the water landing modes, a flashing beacon light, a VHF recovery beacon transmitter, a VHF transceiver, and an H-F transceiver for both landing modes. A recovery loop is also provided on the C/M to facilitate lifting. If the command module enters the water and stabilizes in a stable II (inverted) condition, the uprighting system is activated, inflating two of three air bags (the third air bag is for redundancy in case of a malfunction) causing the command module to assume a stable I (upright) condition. The air bags are then vented allowing them to deflate. If the command module returns to an inverted position prior to recovery, the uprighting system can be reactivated as necessary. The sea (dye) marker and swimmers umbilical are deployed automatically when the recovery antenna is deployed. The marker is tethered to the C/M forward compartment deck and will last approximately 12 hours. The swimmers umbilical provides the electrical connection for communication between the crew in the C/M and the recovery forces during the recovery phase.

3-25. ENVIRONMENTAL CONTROL SYSTEM.

3-26. The basic purpose of the environmental control system (ECS) is to provide a controlled environment for three astronauts in the Apollo spacecraft during missions of up to 14 days duration. This environment consists of a pressurized suit circuit for use during normal or emergency conditions, and a pressurized shirtsleeve cabin atmosphere used only when normal conditions exist. Metabolically, the system is responsible for supplying oxygen and hot and cold potable water; as well as removing carbon dioxide, odors, water production, and heat output. The ECS also disperses electronic equipment heat loads and provides for venting the waste management system (WMS).

3-27. ECS OPERATION.

3-28. The ECS is so designed that a minimum amount of crew time is required for normal system operation. Electrical and mechanical override and backup capabilities exist throughout the system to maintain the required reliability level. Oxygen and potable water are supplied to the ECS by components of the electrical power system (EPS). (See figure 3-6.) The oxygen originates in the cryogenic storage tanks and the potable water is a by-product of the fuel cells, both located in the S/M. Waste water is collected from moisture that condenses within the pressure suit circuit, and is stored for ECS utilization. Block I S/C incorporates additional water tanks in the S/M. This supply permits the accomplishment of maximum duration earth orbital missions.

3-29. In maintaining the pressure suit and cabin shirtsleeve environments, the ECS continually conditions the atmospheres of both. This is accomplished by automatically controlling the supply of oxygen; regulating the flow, pressure, temperature, and humidity; and removing unwanted items such as carbon dioxide, odors, heat, and moisture. The pressure suit and cabin gases are processed for re-use by being routed through the suit circuit debris trap, CO_2-odor absorber filters, and heat exchanger. Additional components installed on Block II S/C permit the CSM ECS to pressurize the LEM.

3-30. A continuous circulating mixture of water-glycol provides the ECS with a heat transport fluid loop. This flow is used to cool the cabin atmosphere, pressure suit circuit, electronic equipment, and a portion of the potable water. It also serves as a heat source for the cabin, when required. All unwanted heat absorbed by the water-glycol is transported to the ECS space radiators located on the surface of the S/M. (These radiators are not to be confused with the EPS radiators also located on this surface.) Should cooling by radiation be inadequate, supplemental cooling takes place within the water-glycol evaporator where heat is rejected by the evaporation of waste water. In addition to this primary coolant loop, Block II S/C are provided with a completely independent, secondary coolant loop. The redundant loop routes water-glycol to certain critical components that are absolutely necessary to complete a safe return to earth.

3-31. The water supply subsystem is concerned with the storage and distribution of potable water produced by the fuel cells, and waste water recovered from the suit heat exchanger. These supplies are used by the ECS to furnish hot and cold potable water for crew consumption, and waste water for evaporative cooling by the water-glycol evaporator and suit heat exchanger.

3-32. ECS ENTRY PROVISION. CSM separation prior to entry removes the capability of cooling the water-glycol by space radiation, and cuts off the primary sources of potable water and oxygen. Provision is therefore made to enable the ECS to carry out its functions following separation. Hours before separation occurs, the C/M cabin is cold-soaked using a flow of cold water-glycol. This provides a heat sink for the aerodynamically developed heat that penetrates the C/M during entry. After separation, the water-glycol and the suit circuit are cooled exclusively by water evaporation. A tank in the C/M supplies all the oxygen required during entry and descent. Upon landing, the postlanding ventilation system is activated.

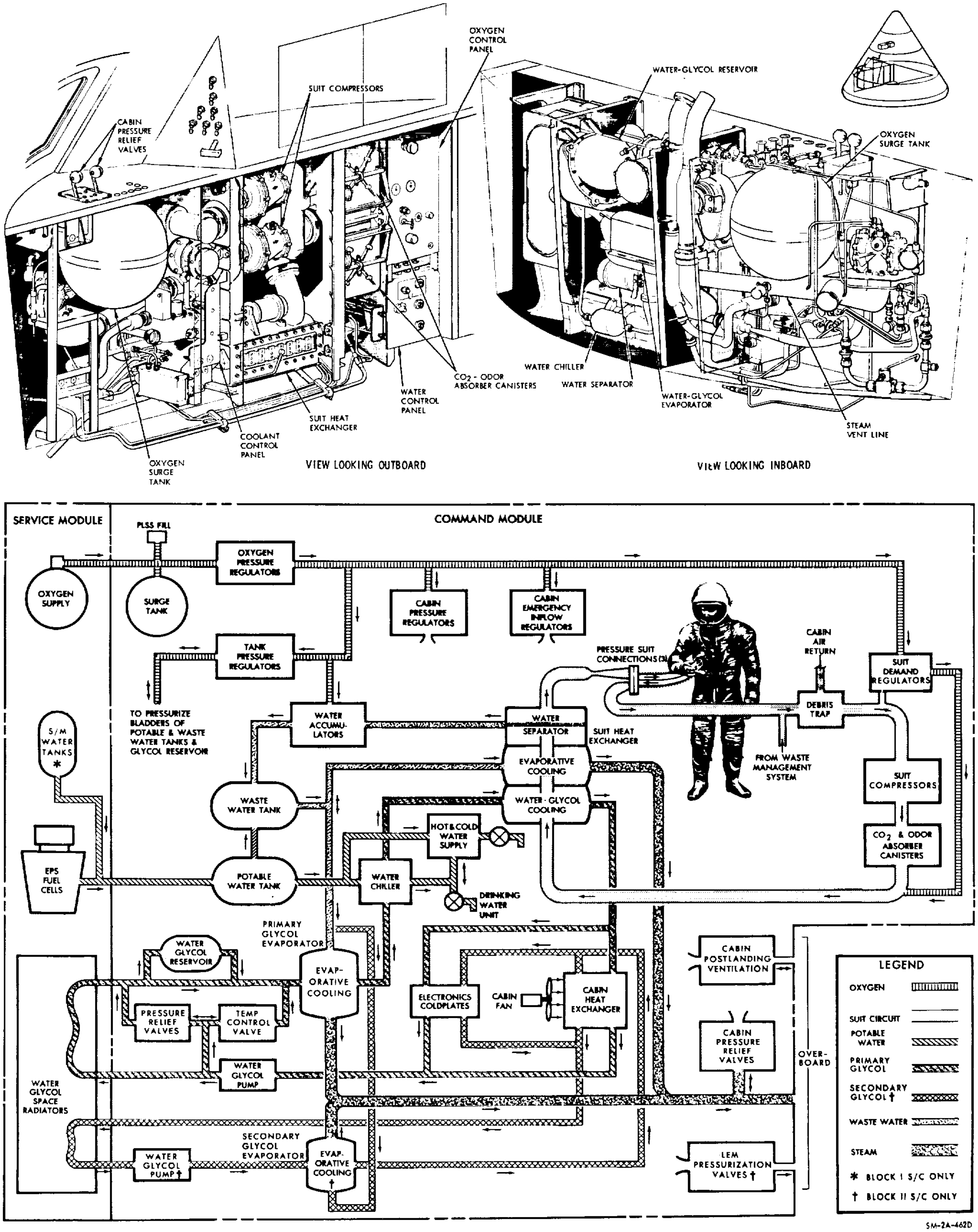


Figure 3-6. Environmental Control System Simplified Flow Diagram

3-33. ELECTRICAL POWER SYSTEM.

3-34. The primary purpose of the electrical power system (EPS) is to provide the electrical energy sources, power generation and controls, power conversion and conditioning, and power distribution to a-c and d-c electrical buses to meet the requirements of the various spacecraft systems during the mission flight and postlanding phases. This is shown in figures 3-7 and 3-9 and described in paragraph 3-43. For ground checkout, all d-c electrical power will be supplied by ground support equipment prior to activation of the fuel cells. During this same period, a-c electrical power will be supplied by the S/C inverters for some checkout functions and by ground support equipment for other checkout functions. The secondary purpose of the EPS is to furnish the environmental control system with potable water required by the three astronauts during the mission. This water is obtained as a by-product of the three fuel cell powerplants. Paragraph 3-43 provides a list of the various power sources and the power-consuming devices connected to each bus.

3-35. D-C POWER SUPPLY.

3-36. Two d-c power sources provide the S/C with 28-volt d-c power. The first source consists of three Bacon-type, hydrox (hydrogen-oxygen) fuel cell powerplants. The second source is obtained from zinc-silver oxide-type storage batteries. The fuel cell powerplants are connected in parallel and used throughout the mission until command-service module (CSM) separation. Any two of the three fuel cells will provide sufficient power for normal mission loads. In the event two fuel cells fail, the third is capable of furnishing emergency power; however, this is contingent upon removing all nonessential loads from the bus, in addition to supplying battery power to the bus at peak loads above the capacity of the one operating fuel cell powerplant. The d-c power system is controlled, regulated, and protected by appropriate switching circuits, undervoltage detection, and circuit breakers. Two nonrechargeable batteries (Block I only), the fuel cell powerplants, the cryogenic storage system for the hydrogen and oxygen, and the glycol coolant space radiators are located in the service module. (See figure 3-8.)

3-37. CRYOGENIC GAS STORAGE SYSTEM. The cryogenic gas storage system supplies the hydrogen and oxygen consumed by the three fuel cell powerplants. In addition, the oxygen used by the ECS is also supplied from this source. The hydrogen and oxygen subsystems are very similar, each consisting of storage tanks, associated valves, pressure switches, motor switches, lines, and other plumbing components. (See figure 3-8.) The hydrogen and oxygen are stored in a super-critical state. However, by the time both reactants reach the fuel cells, they have warmed considerably and are in a gaseous state.

3-38. FUEL CELL POWERPLANTS. Each of the three fuel cell powerplants consist of 31 single cells connected in series. Each cell consists of a hydrogen compartment, an electrolyte compartment, an oxygen compartment, and two electrodes. (See figure 3-8.) The electrolyte is composed of potassium hydroxide (78 percent) and water (22 percent) and remains constant in cell reaction by simply providing an ionic conduction path between the electrodes. The hydrogen electrode is composed of nickel, while the oxygen electrode is composed of nickel and nickel oxide. This electrode structure also remains constant throughout the fuel cell operation. The consumable reactants of hydrogen and oxygen are supplied to the cell under regulated pressure, using nitrogen pressure as a reference as well as for pressurizing the powerplants. By chemical reaction, electricity, water, and heat are produced, with the reactants being consumed in proportion to the electrical load. The by-products, water and heat, are utilized to maintain the supply of potable water and to keep the electrolyte at the proper operating temperature. The 31 fuel cells and the required pumps, valves, regulators, and other components are housed in a container.

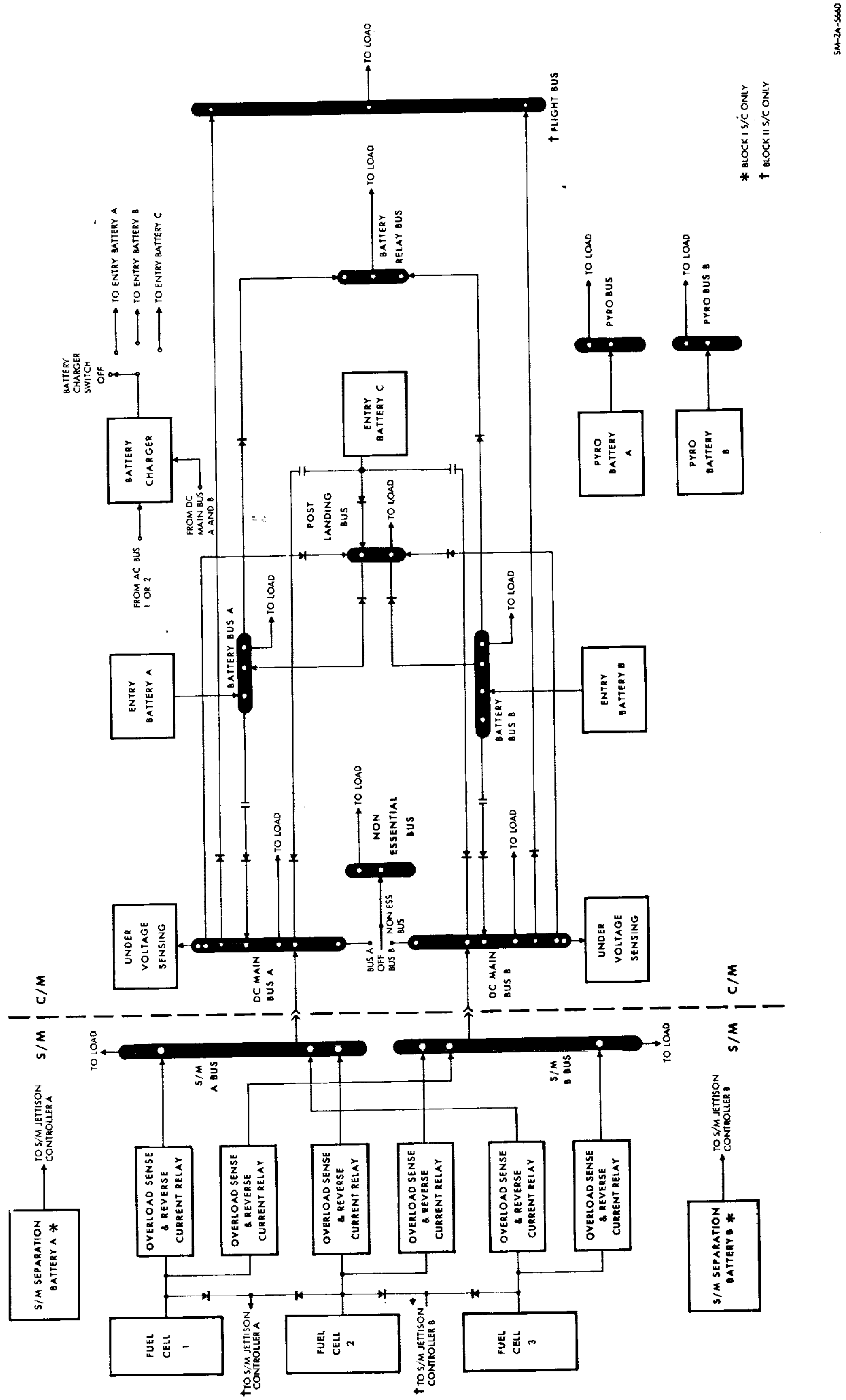


Figure 3-7. Electrical Power System — D-C Power Distribution Diagram

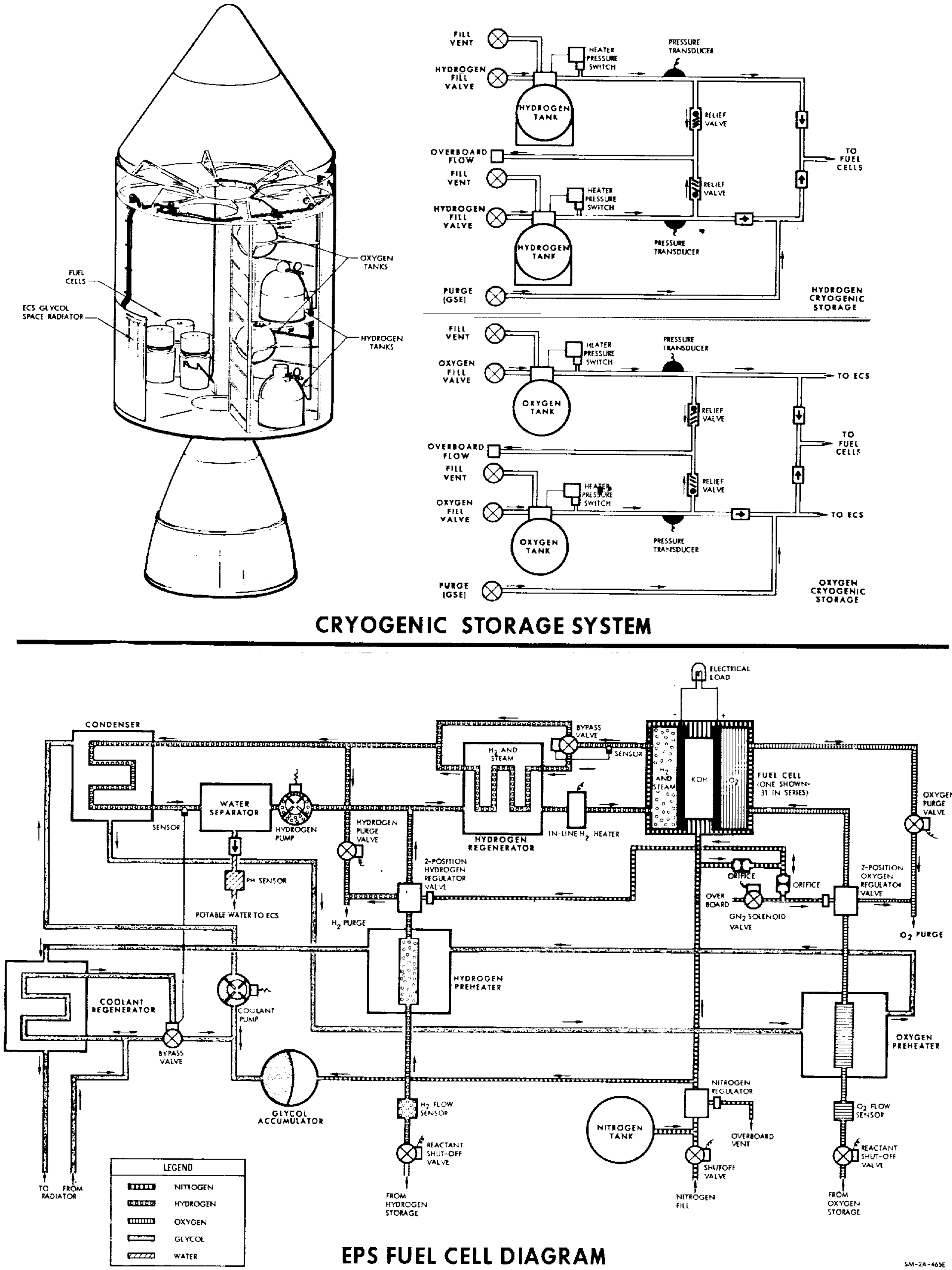


Figure 3-8. Electrical Power System—Cryogenics Storage and Fuel Cell Functional Diagram

3-39. BATTERIES. Three entry batteries, located in the lower equipment bay of the C/M, can be selected and switched to a variety of buses and circuits. The circuitry which ignites the S/C pyrotechnic devices is completely independent and isolated from the remainder of the d-c system, and receives power from two pyrotechnic batteries. There are two nonrechargeable batteries in the S/M (Block I only) whose sole function is to furnish power to the service module jettison controllers. In Block II, S/C power to the S/M jettison controllers is furnished by the fuel cell powerplants. This will sustain the firing of those S/M reaction control engines that provide S/M retrograde following C/M—S/M separation.

3-40. BATTERY CHARGER. A battery charger, located in the lower equipment bay of the C/M, is utilized to assure that the entry batteries are fully charged before entry begins.

3-41. A-C POWER SUPPLY.

3-42. Three solid-state inverters, located in the lower equipment bay of the C/M, are the source of the 115/200-volt 400-cycle 3-phase a-c power used in the S/C. These inverters operate from the two 28-volt d-c main buses, and supply power to two 400-cycle a-c buses. (See figure 3-9.) The a-c electrical power system is complete with adequate switching arrangements, overvoltage and overload sensing circuits, as well as circuit breakers for protection of the various a-c loads. Under normal conditions, one inverter has the capability of supplying all S/C primary 400-cycle a-c electrical power needs. The other two inverters act as redundant standby units. In the event of inverter failure, input and load are manually switched to another inverter. Although the inverters cannot be paralleled due to circuitry provisions, two inverters can operate simultaneously if each supplies a separate a-c bus.

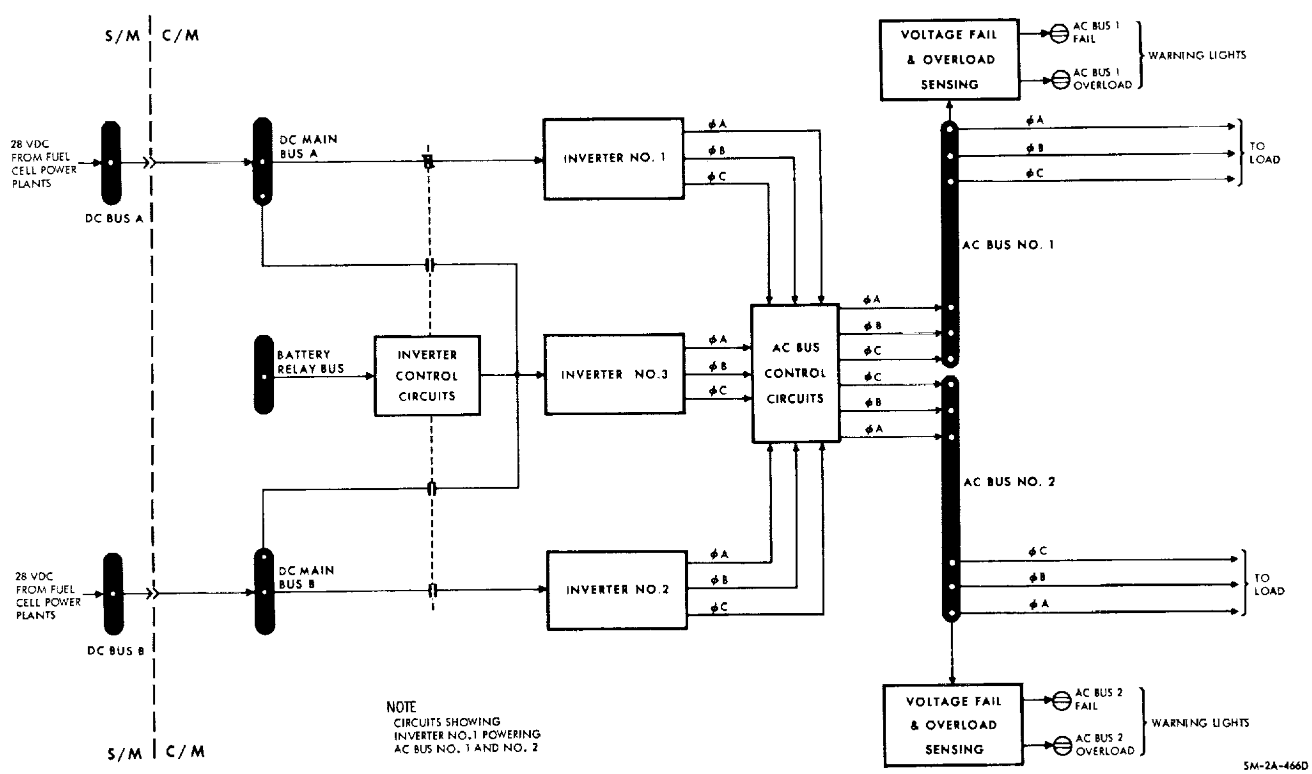


Figure 3-9. Electrical Power System — A-C Power Distribution Diagram

3-43. SPACECRAFT POWER SOURCES AND POWER CONSUMING DEVICES.

3-44. The following list contains Block I and Block II spacecraft power sources and power-consuming devices.

Block I	Block II
Command Module D-C Main Bus A (Powered by fuel cells 1, 2, and 3, and backed up by batteries A, B, and C, when necessary)	
Environmental control system	Environmental control system
Pressure and temperature transducers	Pressure and temperature transducers
Water separator No. 1	Emergency loop temperature transducers
S/M water tank control	Water accumulator
Steam duct heater No. 1	Steam duct heater No. 1 and water-glycol temperature control
Flight and postlanding bus	Compressor inverter
Pyro interrupter switch	Flight and postlanding bus
Oxygen and hydrogen purge - fuel cell powerplants No. 1, 2, and 3	Pyro interrupter switch
Inverters No. 1 and No. 3	Oxygen and hydrogen purge - fuel cell powerplants No. 1, 2, and 3
Battery charger	Inverters No. 1 and No. 3
Nonessential bus switch	Battery charger
Interior floodlighting	Nonessential bus switch
D-C sensing unit and voltmeter switch	Interior floodlighting
Stabilization and control system	D-C sensing unit and voltmeter switch
Direct control	Stabilization and control system
Pitch	Direct control
Roll-channel A&C	Pitch
Roll-channel B&D	Roll-channel A&C
Yaw	Roll-channel B&D
Group 1	Yaw
Group 2	Logic

Block I	Block II
Potable water heater	Potable water heater
Caution and warning detection unit	Caution and warning detection unit
Event timer	Event timer
Central timing equipment	Central timing equipment
Reaction control system	Reaction control system
Gauging	Gauging
Propellant isolation	Propellant isolation
RCS transfer	RCS transfer
RCS heaters	RCS heaters
Essential instrumentation	Essential instrumentation
Cryogenic oxygen and hydrogen tank heaters	Cryogenic oxygen and hydrogen tank heaters
Service propulsion system	Service propulsion system
Gauging	Gauging
Helium shutoff valve	Helium shutoff valve
Guidance and navigation system	Guidance and navigation system
Inertial measurement unit-coupling display unit	Inertial measurement unit-coupling display unit
Inertial measurement unit heaters	Inertial measurement unit heaters
Optics	Optics
Computer	Computer
Space suit communications and biomed instrumentation	Entry monitor display
	Flight bus
Crew couch attenuation	Crew couch attenuation
	Rendezvous radar transponder
	Docking lights

Block I	Block II
Command Module D-C Main Bus B (Powered by fuel cells 1, 2, and 3, and backed up by batteries A, B, and C, when necessary)	
Environmental control system	Environmental control system
Pressure and temperature transducers	Pressure and temperature transducers
Water separator No. 2	Emergency loop temperature transducers
S/M water tank control	Water accumulator
Steam duct heater No. 2	Steam duct heater No. 2 and water-glycol temperature control
Flight and postlanding bus	Flight and postlanding bus
Pyro interrupter switch	Pyro interrupter switch
Oxygen and hydrogen purge - fuel cell powerplants No. 1, 2, and 3	Oxygen and hydrogen purge - fuel cell powerplants No. 1, 2, and 3
Inverters No. 2 and No. 3	Inverters No. 2 and No. 3
Battery charger	Battery charger
Nonessential bus switch	Nonessential bus switch
Interior floodlighting	Interior floodlighting
D-C sensing unit and voltmeter switch	D-C sensing unit and voltmeter switch
Stabilization and control system	Stabilization and control system
Direct control	Direct control
Pitch	Pitch
Roll-channel A&C	Roll-channel A&C
Roll-channel B&D	Roll-channel B&D
Yaw	Yaw
Group 1	Logic
Group 2	
Potable water heater	Potable water heater
Caution and warning detection unit	Caution and warning detection unit

Block I	Block II
Event timer	Event timer
Central timing equipment	Central timing equipment
Reaction control system	Reaction control system
Gauging	Gauging
Propellant isolation	Propellant isolation
RCS transfer	RCS transfer
RCS heaters	RCS heaters
Essential instrumentation	Essential instrumentation
Cryogenic oxygen and hydrogen tank heaters	Cryogenic oxygen and hydrogen tank heaters
Service propulsion system	Service propulsion system
Gauging	Gauging
Helium shutoff valve	Helium shutoff valve
Guidance and navigation system	Guidance and navigation system
Inertial measurement unit-coupling display unit	Inertial measurement unit-coupling display unit
Inertial measurement unit heaters	Inertial measurement unit heaters
Optics	Optics
Computer	Computer
Space suit communications and biomed instrumentation	Entry monitor display
Crew couch attenuation	Crew couch attenuation
	Rendezvous radar transponder
	Docking lights
	LEM power switch

Block I	Block II
Command Module A-C Bus No. 1 (Powered by inverter No. 1, 2, or 3)	
Reactant pump - fuel cell powerplants No. 1, 2, and 3	Reactant pump - fuel cell powerplants No. 1, 2, and 3
Battery charger	Battery charger
Stabilization and control - group 1 and group 2	Stabilization and control - group 1 and group 2
Guidance and navigation system	Guidance and navigation system
Cryogenic fuel quantity amplifier	Cryogenic fuel quantity amplifier
Telecommunications	Telecommunications
Environmental control system	Environmental control system
Glycol pumps	Glycol pumps
Suit compressors	Suit compressors
Cabin air fans, water-glycol temperature control, suit temperature control, and waste management blower	Cabin air fans, water-glycol temperature control, suit temperature control, and waste management blower
	Water-glycol emergency loop
Space radiator isolation valves	Space radiator isolation valves
Interior lighting	Interior lighting
	Exterior lighting
SPS gauging	SPS gauging
A-C sensing unit and voltmeter switch	A-C sensing unit and voltmeter switch
Cryogenic oxygen and hydrogen tank fan motors (system 1)	Cryogenic oxygen and hydrogen tank fan motors (system 1)
Gas analyzer	
Command Module A-C Bus No. 2 (Powered by inverter No. 1, 2, or 3)	
Reactant pump - fuel cell powerplants No. 1, 2, and 3	Reactant pump - fuel cell powerplants No. 1, 2, and 3
Battery charger	Battery charger
Stabilization and control - group 1 and group 2	Stabilization and control - group 1 and group 2

Block I	Block II
Guidance and navigation system	Guidance and navigation system
Cryogenic fuel quantity amplifier	Cryogenic fuel quantity amplifier
Telecommunications	Telecommunications
SPS gauging	SPS gauging
Environmental control system	Environmental control system
Glycol pumps	Glycol pumps
Space radiator isolation valves	Space radiator isolation valves
Cabin air fans, cabin temperature control and water-glycol temperature control	Cabin air fans, cabin temperature control and water-glycol temperature control
Suit compressors	Water-glycol emergency loop
Interior lighting	Suit compressors
A-C sensing unit and voltmeter switch	EVT oxygen valve
Cryogenic oxygen and hydrogen tank fan motors (system 2)	Interior lighting
	Exterior lighting
	A-C sensing unit and voltmeter switch
	Cryogenic oxygen and hydrogen tank fan motors (system 2)
Command Module Battery Bus A (Powered by entry battery A)	
Flight and postlanding bus	Flight and postlanding bus
ELS sequencer A logic and SECS arm	ELS sequencer A logic and SECS arm
Battery charger and battery bus A tie switch	Battery charger and battery bus A tie switch
Arm mission sequencer logic bus and EDS abort enable switch	Battery relay bus
	EDS - bus No. 1
Logic mission sequencer A and voltmeter switch	D-C main bus A
Battery relay bus	D-C sensing unit and voltmeter switch

Block I	Block II
EDS - bus No. 1 D-C main bus A D-C sensing unit and voltmeter switch Main gimbal control - yaw, pitch Uprighting system - compressor No. 1	Main gimbal control - yaw, pitch Uprighting system - compressor No. 1 Flotation bag control
Command Module Battery Bus B (Powered by entry battery B)	
Flight and postlanding bus ELS sequencer B logic and SECS arm Battery charger and battery bus B tie switch Arm mission sequencer logic bus and EDS abort enable switch Logic mission sequencer B and voltmeter switch Battery relay bus EDS - bus No. 3 D-C main bus B D-C sensing unit and voltmeter switch Auxiliary gimbal control - yaw, pitch Uprighting system - compressor No. 2	Flight and postlanding bus ELS sequencer B logic and SECS arm Battery charger and battery bus B tie switch Battery relay bus EDS - bus No. 3 D-C main bus B D-C sensing unit and voltmeter switch Auxiliary gimbal control - yaw, pitch Uprighting system - compressor No. 2 Flotation bag control
Command Module Flight and Postlanding Bus (Powered by entry battery C, d-c main buses A and B, and battery buses A and B)	
VHF recovery beacon D-C main bus A D-C main bus B Audio center (engineer)	D-C main bus A D-C main bus B Microphone amplifiers—NAV, CMDR, ENGR Floodlights

Block I	Block II
Audio center transmitter key relay	ECS postlanding ventilation system
VHF/AM transmitter receiver	Flotation bag No. 3
H-F transceiver	EDS - bus No. 2
Audio center (CMDR)	
Up-data link	
VHF/FM transmitter	
S-band power amplifier	
Unified S-band power relay	
Signal conditioning equipment (Block I)	
TV camera	
C-band transponder	
Data storage equipment	
Premodulation processor	
Audio center (NAV)	
Microphone amplifiers — NAV, CMDR, ENGR	
ECS postlanding ventilation system	
Flotation bag control	
Command Module Battery Relay Bus (Powered by entry batteries A and B)	
Control circuits - inverters No. 1, 2, and 3	Control circuits - inverters No. 1, 2, and 3
A-C buses No. 1 and No. 2 over-undervoltage and overload sensing	A-C buses No. 1 and No. 2 over-undervoltage and overload sensing
Reactant shutoff valves - fuel cell powerplants No. 1, 2, and 3	Reactant shutoff valves - fuel cell powerplants No. 1, 2, and 3
D-C main buses A and B undervoltage sensing unit	D-C main buses A and B undervoltage sensing unit

Block I	Block II
D-C main buses A and B select switch, fuel cell powerplants No. 1, 2, and 3	D-C main buses A and B select switch, fuel cell powerplants No. 1, 2, and 3 Fuel cell powerplants No. 1, 2, and 3 radiator valves
Command Module Nonessential Bus (Powered by d-c main bus A or B)	
Nonessential instrumentation	Nonessential instrumentation
NASA scientific instrumentation	NASA scientific instrumentation
Flight qualification recorder	Special equipment bays No. 1 and No. 2
Special equipment bays No. 1 and No. 2 (S/C 012 and S/C 014)	Special equipment hatch
Special equipment hatch (S/C 012 and S/C 014)	
Command Module MESC Pyro Bus A (Powered by pyro battery A)	
Sequencer A	Sequencer A
RCS fuel dump and voltmeter switch	HF orbital antenna deploy
LES, ELS, and RCS pressure initiators	RCS fuel dump and voltmeter switch
	LES, ELS, and RCS pressure initiators
Command Module MESC Pyro Bus B (Powered by pyro battery B)	
Sequencer B	Sequencer B
RCS fuel dump and voltmeter switch	HF recovery antenna deploy
LES, ELS, and RCS pressure initiators	RCS fuel dump and voltmeter switch
	LES, ELS, and RCS pressure initiators
Command Module Entry Battery C	
D-C main bus A	D-C main bus A
D-C main bus B	D-C main bus B
Flight and postlanding bus	Flight and postlanding bus
EDS - bus No. 2	Voltmeter switch
Voltmeter switch	

Block I	Block II
Service Module D-C Bus A (Powered by fuel cells 1, 2, and 3)	
SPS gimbal motor Overload or reverse current sensing - fuel cell powerplants No. 1, 2, and 3 D-C main bus A	SPS gimbal motor Overload or reverse current sensing - fuel cell powerplants No. 1, 2, and 3 S/M jettison controller A D-C main bus A
Service Module D-C Bus B (Powered by fuel cell 1, 2, or 3)	
SPS gimbal motor Overload or reverse current sensing - fuel cell powerplants No. 1, 2, and 3 D-C main bus B	SPS gimbal motor Overload or reverse current sensing - fuel cell powerplants No. 1, 2, and 3 S/M jettison controller B D-C main bus B
Service Module Jettison Controller Battery A	
Controller A	None
Service Module Jettison Controller Battery B	
Controller B	None
Flight Bus (Powered by d-c main buses A and B)	
None	Rendezvous radar transponder S-band PA No. 1 transponder S-band power amplifier No. 2 Up-data link Signal conditioning equipment Premodulation processor Data storage and S-band transmitter 2-KMC high-gain antenna

3-45. REACTION CONTROL SYSTEM.

3-46. The reaction control system (RCS) is comprised of two subsystems: the service module and the command module reaction control systems. (See figures 3-10 and 3-11.) The primary purpose of each subsystem is to provide propulsion impulses, as required, for the accomplishment of normal and emergency attitude maneuvers of the spacecraft or the C/M. Both subsystems operate in response to automatic control signals originating from the G&N or stabilization and control system. Manual control is provided by the crew rotation hand controllers. The subsystems are similar to the extent that both utilize pressure-fed, hypergolic propellants, and maintain total redundancy of critical components and rocket engine thrust vectors.

3-47. SERVICE MODULE REACTION CONTROL.

3-48. The S/M-RCS consists of four independent, equally capable, and functionally identical packages, as shown in figure 3-10. Each package contains four reaction control engines, fuel and oxidizer tanks, a helium tank, and associated components such as regulators, valves, filters, lines, and a nucleonic quantity gauging system. These components are mounted on a panel, or package that is installed on the exterior of the S/M near the forward end. All components, with the exception of the engines, are located inside the S/M. In each package, two of the engines are for roll control, and the other two are for pitch or yaw control, depending upon the location of the package. Hypergolic propellants for the S/M-RCS consist of a 50:50 blend of UDMH and hydrazine as fuel and nitrogen tetroxide as oxidizer.

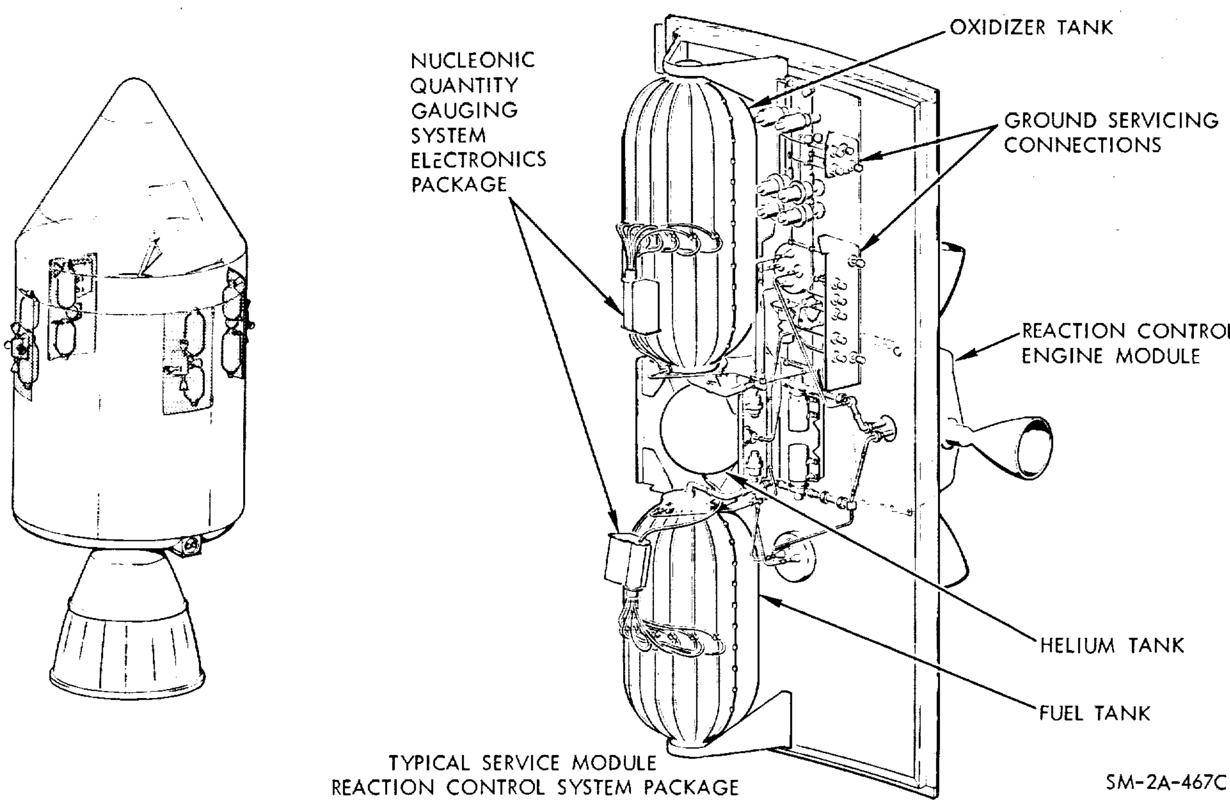


Figure 3-10. Service Module Reaction Control System (Sheet 1 of 2)

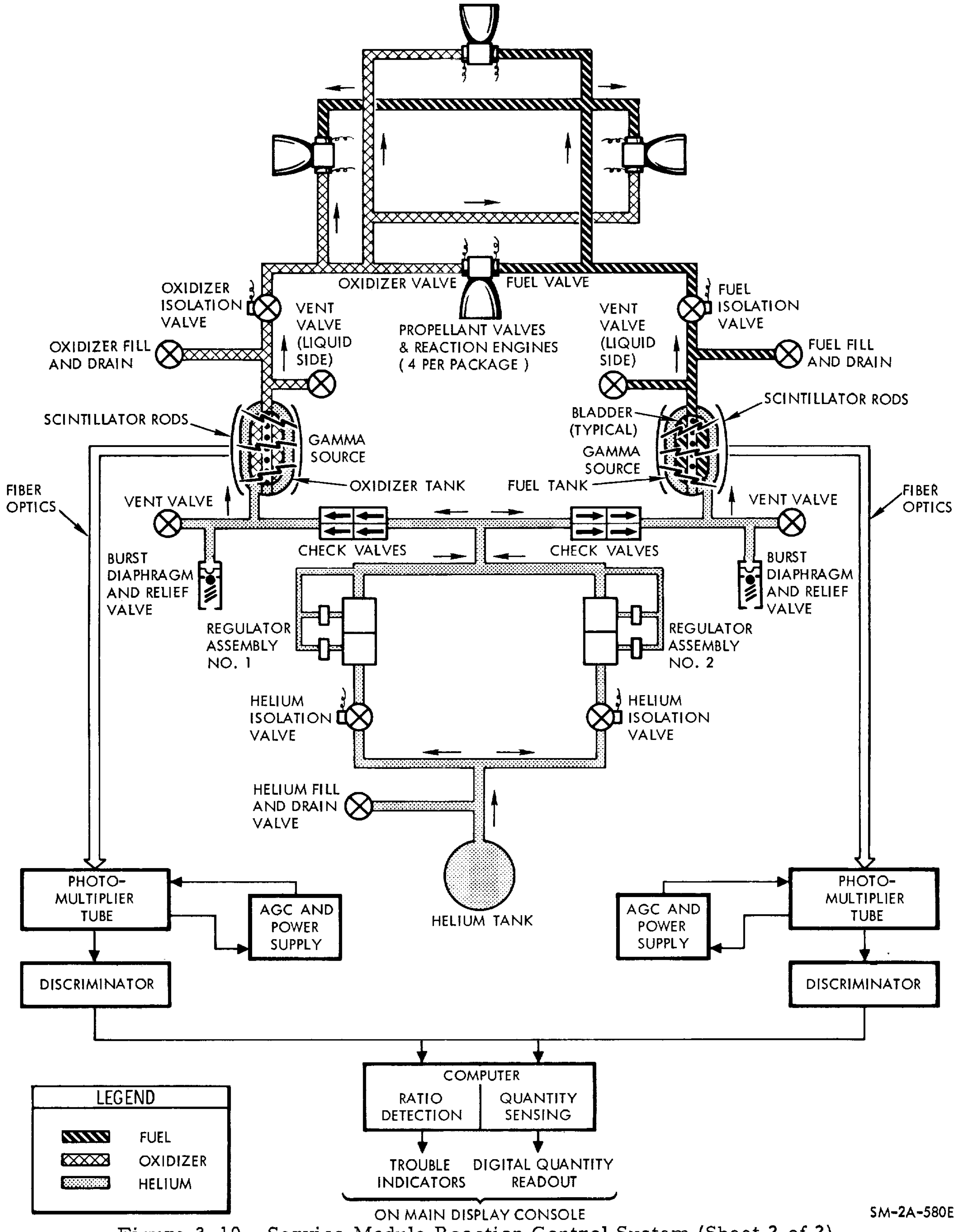


Figure 3-10. Service Module Reaction Control System (Sheet 2 of 2)

3-49. During an Apollo 14-day lunar mission, the S/M-RCS will be used to accomplish many of the following maneuvers: the service propulsion system ullage maneuver, thrust vectors for three-axis stabilization and attitude control, separation of various combinations of modules and/or boosters under normal or abort conditions, LEM docking and separation, and minor orbital or midcourse velocity corrections.

3-50. COMMAND MODULE REACTION CONTROL.

3-51. The C/M-RCS, although similar to the S/M subsystem, is different in several respects including propellant distribution to the reaction control engines. (See figure 3-11.) The C/M contains two independent, equally capable, and functionally identical reaction control systems. Each system consists of two reaction control engines per axis (pitch, roll, and yaw), propellant storage and pressurization tanks, and associated components and lines. Hypergolic propellants for the C/M-RCS consist of nitrogen tetroxide as oxidizer, and monomethyl-hydrazine as fuel. All of the components of this subsystem are located in the aft compartment with the exception of the two negative pitch engines which are located in the forward compartment. For thrust in the pitch and yaw axes, engines from each system are mounted in pairs; whereas for roll, each pair of engines consist of a thrust left and a thrust right engine. In either case, the presence of the second system provides the required redundancy. The C/M-RCS is not activated until CSM separation takes place. The RCS is used to provide attitude maneuver capabilities during entry and, in the event of a high-altitude abort after launch escape system jettison, to provide three-axis control to stabilize C/M motion prior to deployment of the ELS parachutes. Due to the fact that no hypergolic propellant should be onboard the C/M at the time of earth impact, certain provisions are necessary that are not included in the S/M-RCS. Additional components, including squib-operated valves, accomplish burning of the fuel and oxidizer remaining after entry or high-altitude abort, or dump the total propellant load after a pad or a low-altitude abort. Following either of these operations, other squib valves are activated to allow complete helium purging of the C/M-RCS fluid lines and engines.

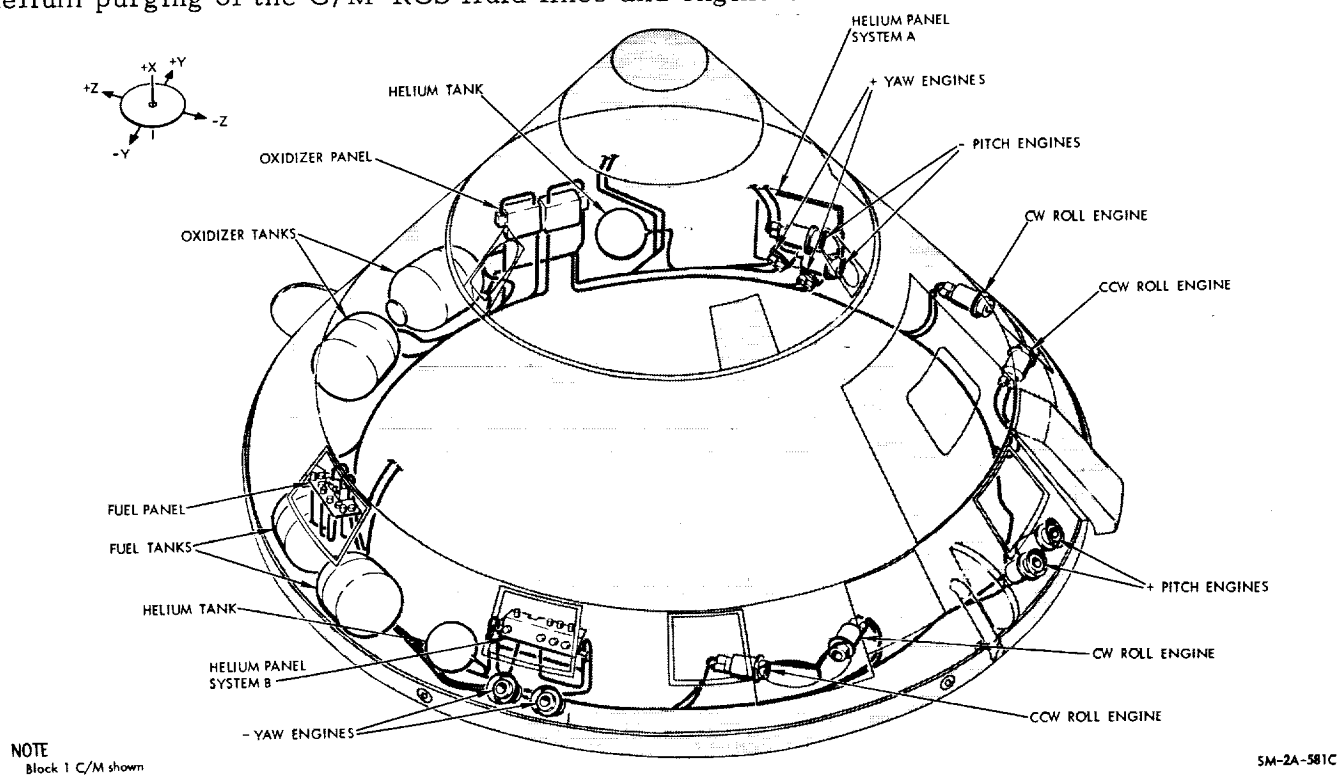


Figure 3-11. Command Module Reaction Control System (Sheet 1 of 2)

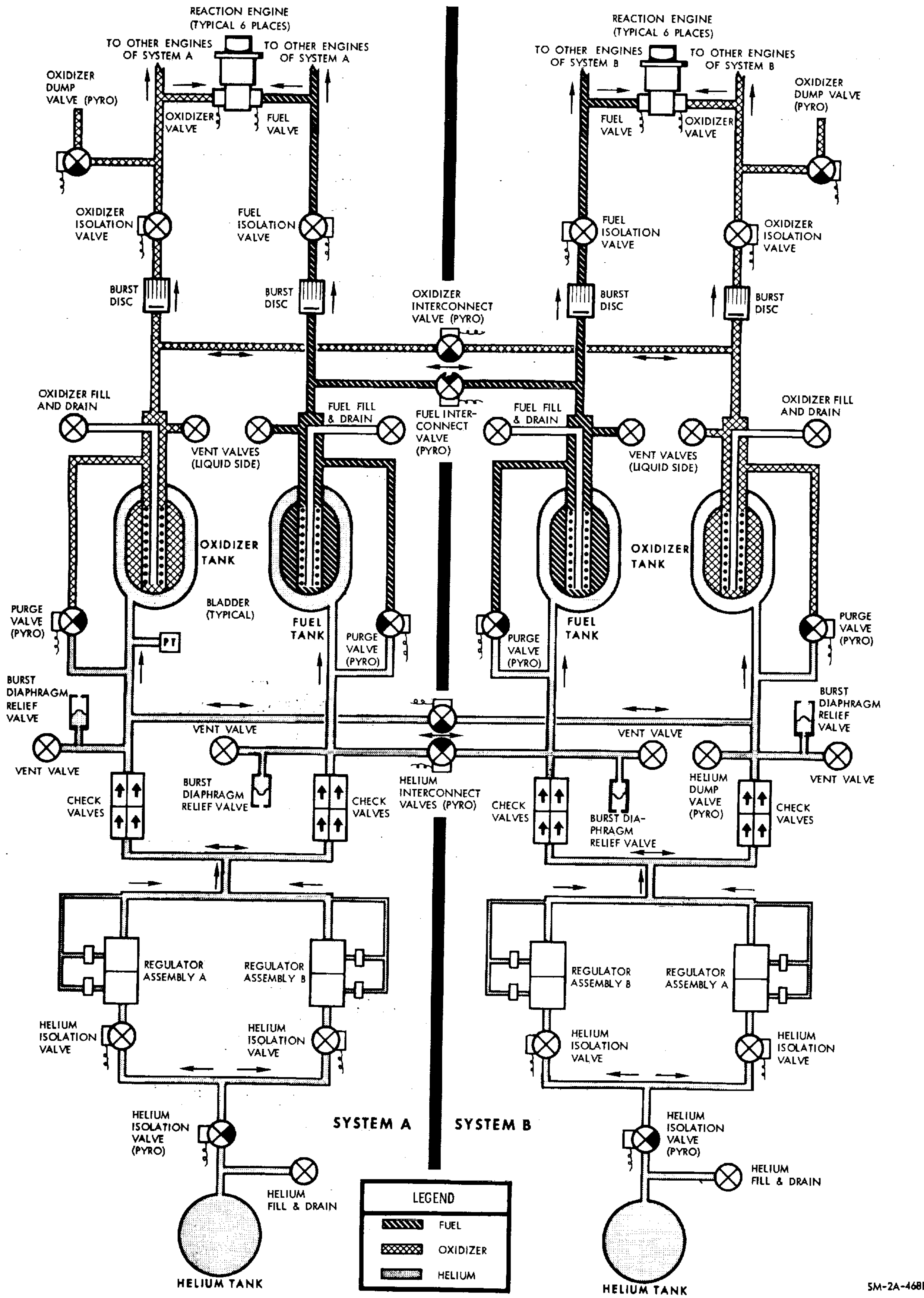


Figure 3-11. Command Module Reaction Control System (Sheet 2 of 2)

3-52. SERVICE PROPULSION SYSTEM.

3-53. The service propulsion system (SPS) provides the thrust required for large changes in spacecraft velocity after booster separation. The SPS consists of a gimbal-mounted single-rocket engine, pressurization and propellant tanks, and associated components, all of which are located in the service module. (See figure 3-12.) Conditions and time will vary the use of the SPS thrust. During a lunar landing mission, for example, the SPS could conceivably be fired for many events. The first might occur shortly after booster separation to carry out an abort during the post-atmospheric portion of the launch trajectory. It could also be used for earth orbit injection or transferring from one earth orbit to another. Following translunar injection, normal midcourse corrections or a post-injection abort could be accomplished using the SPS. Further along on the mission, the SPS provides for insertion of the S/C into lunar orbit, as well as ejection from lunar orbit (transearth injection) into a transearth trajectory. During the lunar orbit phase, transferral from one orbit to another is also possible.

3-54. Hypergolic propellants for the SPS consist of a 50:50 blend of UDMN and hydrazine as fuel, and nitrogen tetroxide as oxidizer. The storage and distribution system for these propellants consist of two fuel tanks, two oxidizer tanks, two helium tanks, associated components, lines, and electrical wiring. Pressurization of the propellant tanks is accomplished using helium. Automatic control and regulation is utilized in the SPS, as well as backup components and operational modes. A quantity gauging system is provided for monitoring the amount of propellant remaining in the tanks.

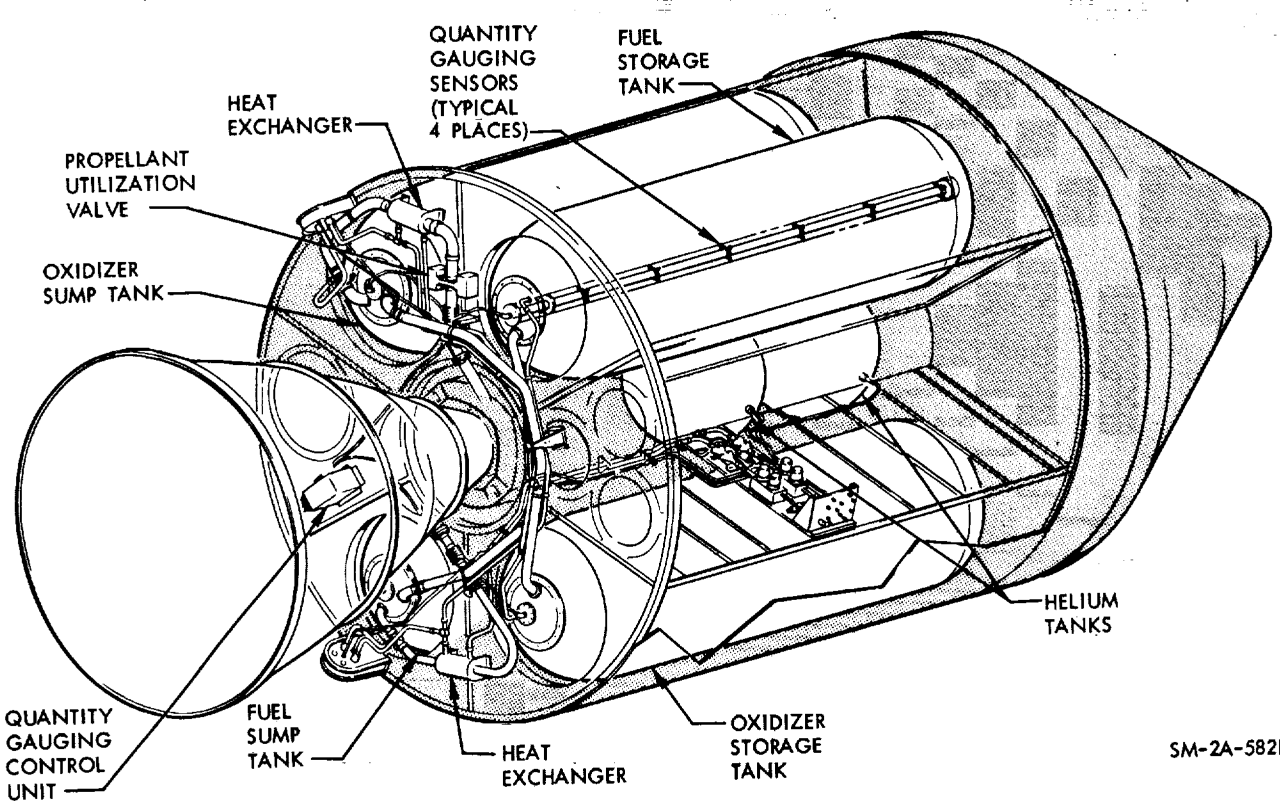


Figure 3-12. Service Propulsion System (Sheet 1 of 2)

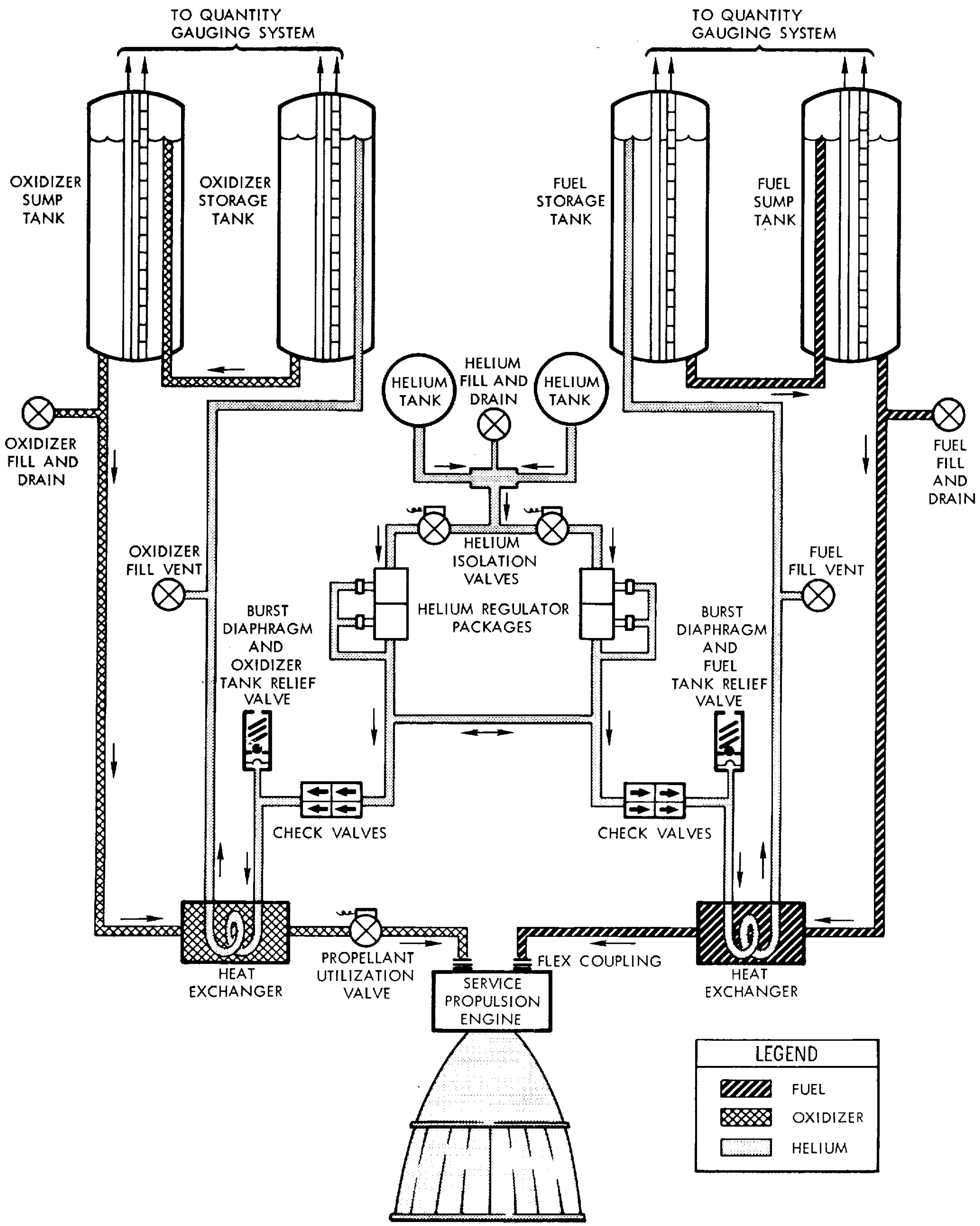


Figure 3-12. Service Propulsion System (Sheet 2 of 2)

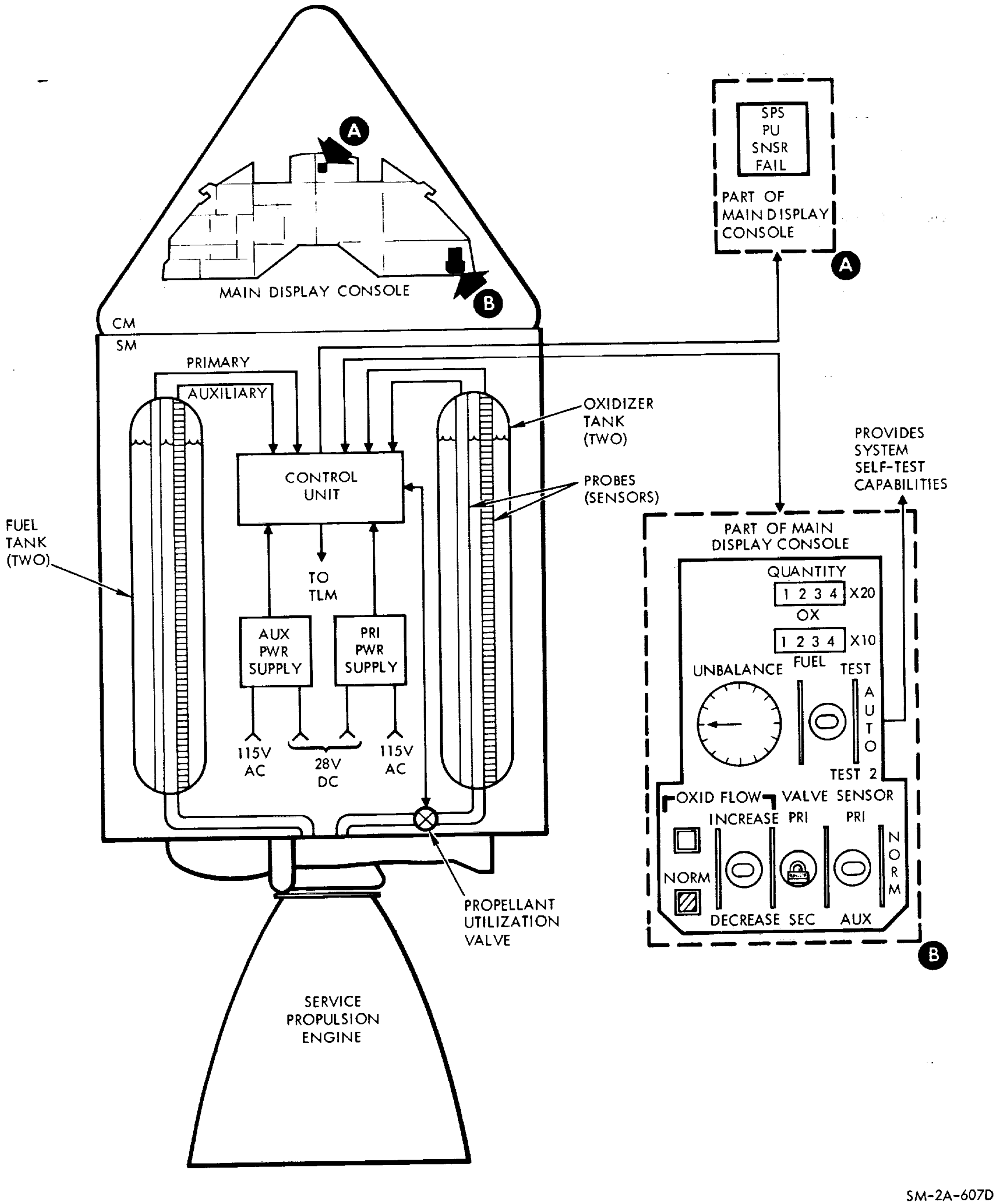


Figure 3-13. SPS Quantity Gauging and Propellant Utilization Systems—Block Diagram

3-55. SERVICE PROPULSION SYSTEM OPERATION.

3-56. The operation of the SPS is in response to an automatic firing command generated by the G&N system, or manual initiation by the crew. The engine assembly is gimbal-mounted to allow engine thrust-vector alignment with the S/C center of gravity to preclude S/C tumbling. Thrust-vector alignment control is maintained automatically by the SCS or manually by the crew utilizing the translation hand control. The control and monitoring of the SPS from the C/M is the only point of SPS interface with the C/M. The engine has no throttle, thus it produces a single value of thrust for velocity increments. It has the capability of restarting a sufficient number of times to complete a 14-day mission.

3-57. SPS QUANTITY GAUGING AND PROPELLANT UTILIZATION SYSTEM. Service propulsion system propellant quantities are monitored and controlled by the system illustrated in figure 3-13. Propellant quantity sensing incorporates two separate systems: primary and auxiliary. The primary quantity sensors are cylindrical capacitance probes mounted axially in each tank. The auxiliary network utilizes impedance-type point sensors providing a step function impedance change when the liquid level passes their location centerline. Auxiliary system electronics provide time integration to permit a continuous measurement when the propellant level is between point sensors.

3-58. Sensor outputs are applied to a control unit containing servo loops which provide output signals representative of the propellant quantity. Two primary fuel servos and one auxiliary fuel servo provide an input signal to the fuel display servo, which, in turn, positions the digital fuel quantity display on the main display console. Two primary oxidizer servos and one auxiliary oxidizer servo provide an input to the oxidizer display servo, which, in turn, positions the digital oxidizer quantity display. The primary and auxiliary fuel and oxidizer servo outputs are also applied to the unbalance display servo, which will sense any unbalance in the remaining fuel-oxidizer quantities and display the amount of unbalance on the unbalance display dial. The propellant utilization valve assembly, installed in the oxidizer engine feed line, incorporates two identical, motor-operated gates to provide redundant oxidizer flow rate control. The two gates (one primary and one auxiliary) are operated manually; to allow oxidizer flow rates to be increased or decreased, to compensate for an unbalance condition in the oxidizer-fuel ratio, and to insure simultaneous propellant depletion. Gate position is controlled by switches on the main display console. The valve incorporates a position potentiometer and the output is applied to a valve position display servo which positions an oxidizer flow indicator. Propellant quantity signals are also routed to telemetry. Quantity and unbalance signals are monitored by the discrepancy warning lights which will provide an alarm in the event of an excessive propellant unbalance condition. A self-test system is incorporated which provides an operational check of the sensing system voltages to the different servos, electronics, and display readings. Self-tests are initiated manually by a test switch on the main display console.

3-59. <u>GUIDANCE AND NAVIGATION SYSTEM.</u>

3-60. The guidance and navigation (G&N) system is a semi-automatic system, directed and operated by the flight crew, which performs two basic functions: inertial guidance and optical navigation. The system consists of inertial, optical, and computer subsystems, each of which can be operated independently, if necessary. Thus a failure in one subsystem will not disable the entire system. The G&N equipment is located in the lower equipment bay and on the main display console of the command module. (See figure 3-14.)

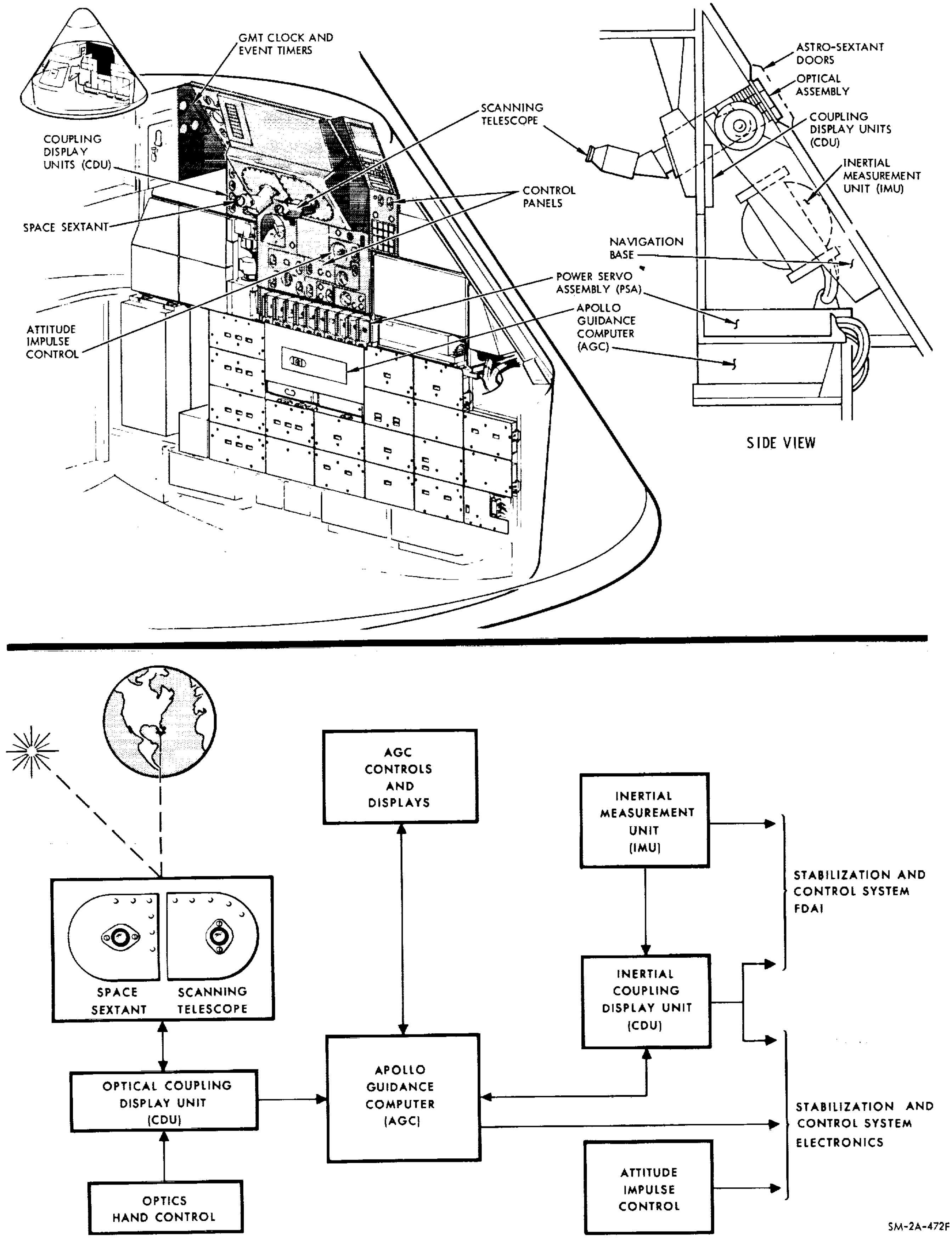


Figure 3-14. Guidance and Navigation System (Block I)

3-61. The three subsystems, individually or in combination, can perform the following functions:

a. Periodically establish an inertial reference which is used for measurements and computations.

b. Align the inertial reference by precise optical sightings.

c. Calculate the position and velocity of the spacecraft by optical navigation and inertial guidance.

d. Generate steering signal and thrust commands necessary to maintain the required S/C trajectory.

e. Provide the flight crew with a display of data which indicates the status of the G&N problem.

3-62. The inertial subsystem consists of an inertial measurement unit (IMU), associated hardware, and appropriate controls and displays. Its major functions involve: (1) measuring changes in S/C attitude, (2) assisting in the generation of steering commands for the S/C stabilization and control system (SCS), and (3) measuring S/C velocity changes due to thrust. Various subsystem modes of operation can be initiated automatically by the computer subsystem or manually by the flight crew, either directly or though appropriate programing of the computer subsystem.

3-63. The optical subsystem consists of a scanning telescope, a sextant, associated hardware, and appropriate controls and displays. Its major functions involve: (1) providing the computer subsystem with data obtained by measuring angles between lines of sight to celestial objects, and (2) providing measurements for establishing the S/C inertial reference. The scanning telescope and sextant are used by the flight crew to take sightings on celestial bodies, landmarks, and the LEM subsequent to separation and during rendezvous. These sightings, when used in conjunction with a catalog of celestial objects stored in the computer subsystem, enable determination of the S/C position and orientation in space. Communication with ground tracking stations provides primary navigation information. The identity of celestial objects and the schedule of measurements is based on an optimum plan determined prior to launch.

3-64. The computer subsystem consists of an Apollo guidance computer (AGC) and appropriate controls and displays. Its major functions involve: (1) calculating steering signals and discrete thrust commands to keep the S/C on a desired trajectory, (2) positioning the IMU stable platform to an inertial reference defined by optical measurements, (3) performing limited G&N system malfunction isolation, and (4) supplying pertinent S/C condition information to appropriate display panels. The AGC is a general purpose digital computer employing a core memory, parallel operation, and a built-in self-check capability. Programs are stored in the AGC and manually or automatically selected to control and solve flight equations. Using information from navigation fixes, the AGC computes a desired trajectory and calculates necessary corrective attitude and thrust commands. Velocity corrections are measured by the inertial subsystem and controlled by the computer subsystem. Velocity corrections are not made continuously but are initiated at predetermined checkpoints in the flight to conserve rocket propellants. The G&N, SCS, SPS, and RCS systems combine to provide closed-loop control of the S/C velocity and attitude.

3-65. STABILIZATION AND CONTROL SYSTEM.

3-66. The stabilization and control system (SCS) for Block I S/C provides control and monitoring of the spacecraft attitude and rate control of the thrust vector of the service propulsion engine, and a backup inertial reference system. (See figure 3-15.) The system may be operated automatically or manually in various modes. The guidance and

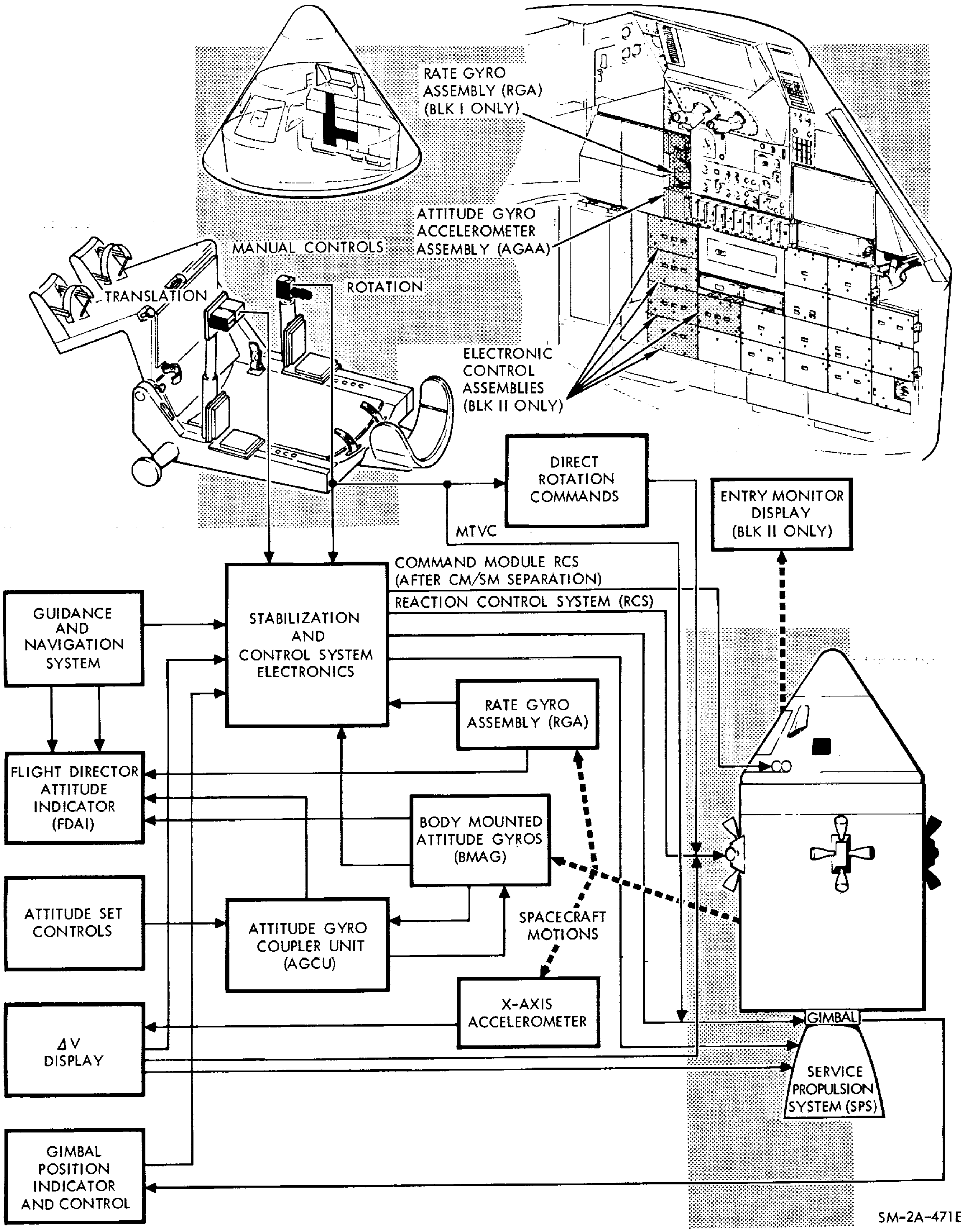


Figure 3-15. Stabilization and Control System

navigation system, service propulsion system, and the CSM reaction control system interface with the SCS. The major components of the SCS, all located in the C/M are: rate gyro assembly; attitude gyro/accelerometer assembly; pitch, yaw, and roll electronic control assemblies (ECAs); display/attitude gyro accelerometer assembly ECA; auxiliary ECA, velocity change indicator, gimbal position/attitude set indicator, flight director attitude indicator (FDAI), two rotation controls, and two translation controls. System controls and displays are located on the C/M main display console. The rate gyro assembly consists of three rate gyros mounted mutually 90 degrees apart in X-, Y-, and Z-axes. The rate gyros provide signals representative of S/C attitude change rates. The rate is displayed on the FDAI and is used by the SCS for damping and stabilization. The attitude gyro accelerometer assembly consists of three body-mounted gyros (BMAGs), and a pendulous accelerometer mounted coincident with the X-axis. The BMAGs sense pitch-, yaw-, and roll-attitude changes and provide attitude-error signals to the FDAI for display, and to the SCS for attitude control. The accelerometer provides acceleration data for automatic termination of SPS thrusting and for display on the ΔV REMAINING indicator. The ECAs are electronic modules which process and condition the input and output electrical signals of the SCS components.

3-67. STABILIZATION AND CONTROL SYSTEM OPERATION.

3-68. The SCS may be used in any one of eight modes, which are selectable by the crew. The SCS attitude control mode automatically maintains S/C attitude within the minimum or maximum deadband limits, which are approximately ±0.5 degrees or ±5.0 degrees respectively. When the S/C exceeds the selected deadband limits, attitude-error signals generated by the BMAGs are applied to circuitry within the ECAs which initiates firing of the proper RCS engines to return the S/C within the selected deadband limits. During G&N attitude control mode, the attitude error consists of the difference between the inertial measurement unit (IMU) and coupling display unit (CDU) output signals. The SCS local vertical mode operation is identical to the SCS attitude control mode with one exception. During local vertical mode operation, an orbit rate signal is applied to the attitude gyro coupling unit to maintain local vertical reference with respect to the earth. The G&N ΔV mode is the primary mode when S/C velocity changes are required. The SPS thrust on-off signal is applied to the SCS electronics by the Apollo guidance computer (AGC). In the G&N mode, automatic thrust vector control (TVC) is accomplished with the G&N attitude error signals, the SCS rate gyro signals, and the TVC electronics. In the SCS ΔV mode, SPS thrust-on signal is initiated manually by the crew. Thrust vector control is accomplished with the attitude error signals generated by the BMAGs, the SCS rate gyro signals, and the TVC electronics. Termination of SPS thrusting is automatic and occurs when the X-axis accelerometer senses that the desired velocity change and the ΔV REMAINING indicator reads zero. Thrust on or off may be accomplished manually if automatic on or off functions fail to occur. Manual control of the SPS thrust vector with rate damping is available to the crew. The translation control is rotated CW for manual control (MTVC) and the rotation control is used to command the SPS engine in the pitch and yaw axis. During entry the G&N entry mode is the primary mode. After CSM separation, the G&N system provides automatic control of the S/C lift vector to effect a safe entry trajectory. In the SCS entry mode, the crew is required to manually control the S/C lift vector with the rotation controller. The SCS is considered a backup system during ΔV maneuvers and entry. The monitor mode permits the crew to monitor S/C attitude, attitude error, and attitude change rate on the FDAI during ascent, and provides rate stabilization after S-IVB separation.

3-69. SPACECRAFT CONTROL PROGRAMERS.

3-70. CONTROL PROGRAMER.

3-71. The control programer (M1), installed in S/C 009, automatically controls the stabilization control system (SCS) and mission sequencer, and conditions ground command signals. The control programer consists of a timer assembly, radio command control assembly, and automatic command control assembly. Equipment required with the control programer consists of a backup attitude reference system (ARS), and radio command equipment. Redundancy is provided to preclude the possibility of losing a C/M due to a single failure. A functional block diagram of the control programer is shown in figure 3-16; the mission description for S/C 009 is presented in section VII.

3-72. MISSION CONTROL PROGRAMER.

3-73. The mission control programer (M3), installed in S/C 006, 011, 017, and 020 programs spacecraft functions which are commanded by the G&N system, S-IVB instrument unit, and the up-data link. The guidance and control (G&C) functions are controlled by the G&N system. The S/C will have operational controls and displays. Characteristics incorporated in the mission control programer include relay switching of automated functions (relay logic) and fixed time-delay relays as required, minimum interference with S/C control and display operation, real time of control of functions computed for C/M recovery, and redundant critical functions. Another model of the programer (M2) will be used in S/C 012 and 014. A functional block diagram of the mission control programer is shown in figure 3-16; the mission description for S/C 011 is presented in section VII.

3-74. SPACECRAFTS 009 and 011 PROGRAMER COMPARISON.

3-75. The following list provides a functional comparison of the control programers used in S/C 009 and S/C 011:

Spacecrafts 009 and 011 Programer Comparison

Functions	Control Programer (S/C 009)	Mission Control Programer (S/C 011)
Mission director	Primary	None
S/C system sequencing	Primary	Initiated by G&N and S-IVB instrument unit
Attitude reference	SCS (Primary) and attitude reference system (backup)	G&N (Primary) and SCS (backup)
Corrective action	EPS only	Limited
Ground control	G&C and staging	Attitude control and staging
Abort capability (maneuvers)	Self-contained and ground control	Ground control
Displays and controls	Partial	Complete and operational

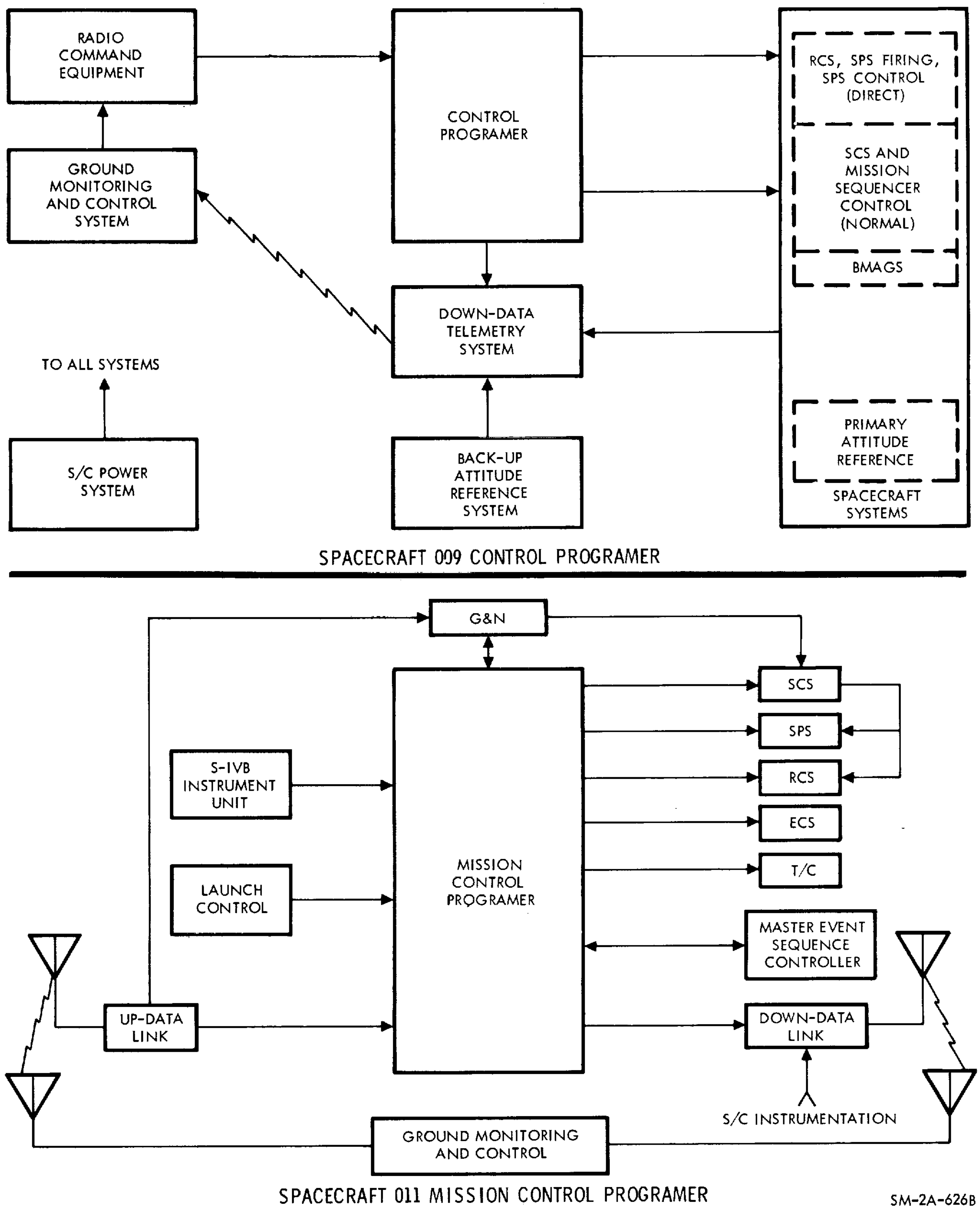


Figure 3-16. Control Programer Functional Block Diagram for S/C 009 and S/C 011

3-76. <u>CREW SYSTEM.</u>

3-77. The purpose of the crew system is to provide for needs peculiar to the presence of the crew aboard the spacecraft. Crew system equipment includes certain degrees of physical protection against radiation, acceleration, impact, and sustained weightlessness. Equipment and provisions for the routine functions of eating, drinking, sleeping, body cleansing, and the elimination of waste are part of the crew system, as is the survival equipment which is provided for abnormal or emergency conditions.

3-78. CREW COUCHES.

3-79. The command module is equipped with three couches, each with adjustable headrests, restraint harness assemblies, and foot-strap restraints. (See figure 3-17.) The basic support for all couches is a fixed frame suspended from shock attenuators. Angular adjustments for hips permit individual comfort during all flight modes. The attenuators lessen the impact forces imposed on the crew during C/M touchdown on water or land.

3-80. PERSONAL EQUIPMENT.

3-81. Each astronaut will have personal equipment available to him during the course of a 14-day mission. The equipment includes a constant-wear communications assembly, a constant-wear garment, a pressure garment assembly (pressure suit), an umbilical assembly, a bioinstrument accessories kit, radiation dosimeters, an emergency medical kit, and a physiological clinical monitoring instrument set. In addition, a thermal insulation overgarment and a portable life support system (PLSS) will be included with the Block II spacecraft.

3-82. The constant-wear communications assembly provides communications between crew members and with MSFN. The assembly consists of a head protection helmet with two microphones and two earphones attached, and will be used in a shirtsleeve environment.

3-83. The constant-wear garment (CWG) provides the astronauts with a basic garment to be worn at all times during a mission. The CWG consists of a two-piece long-sleeved, close-fitting garment which covers the crewman's entire body with the exception of the hands and head.

3-84. The thermal insulation overgarment coverall is an overgarment which completely covers the pressure garment and PLSS. It provides thermal and radiation protection for the astronauts during extravehicular operations.

3-85. The pressure garment assembly (PGA) consists of a torso-and-limbs covering, integral boots, gloves, and helmet, and covers the entire body of an astronaut. The garment is air-tight to support life, in conjunction with the ECS or PLSS, during adverse mission conditions and lunar exploration.

3-86. The umbilical assembly provides the interface between the S/C systems and the PGA. The electrical link between the astronauts and the S/C communications equipment is provided by the electrical umbilical. The oxygen umbilical (plus two umbilical nozzles) provides a means of transferring oxygen from the ECS to the PGA. Another hose assembly, provided on Block II manned S/C, is used to transfer oxygen from the ECS to the PLSS.

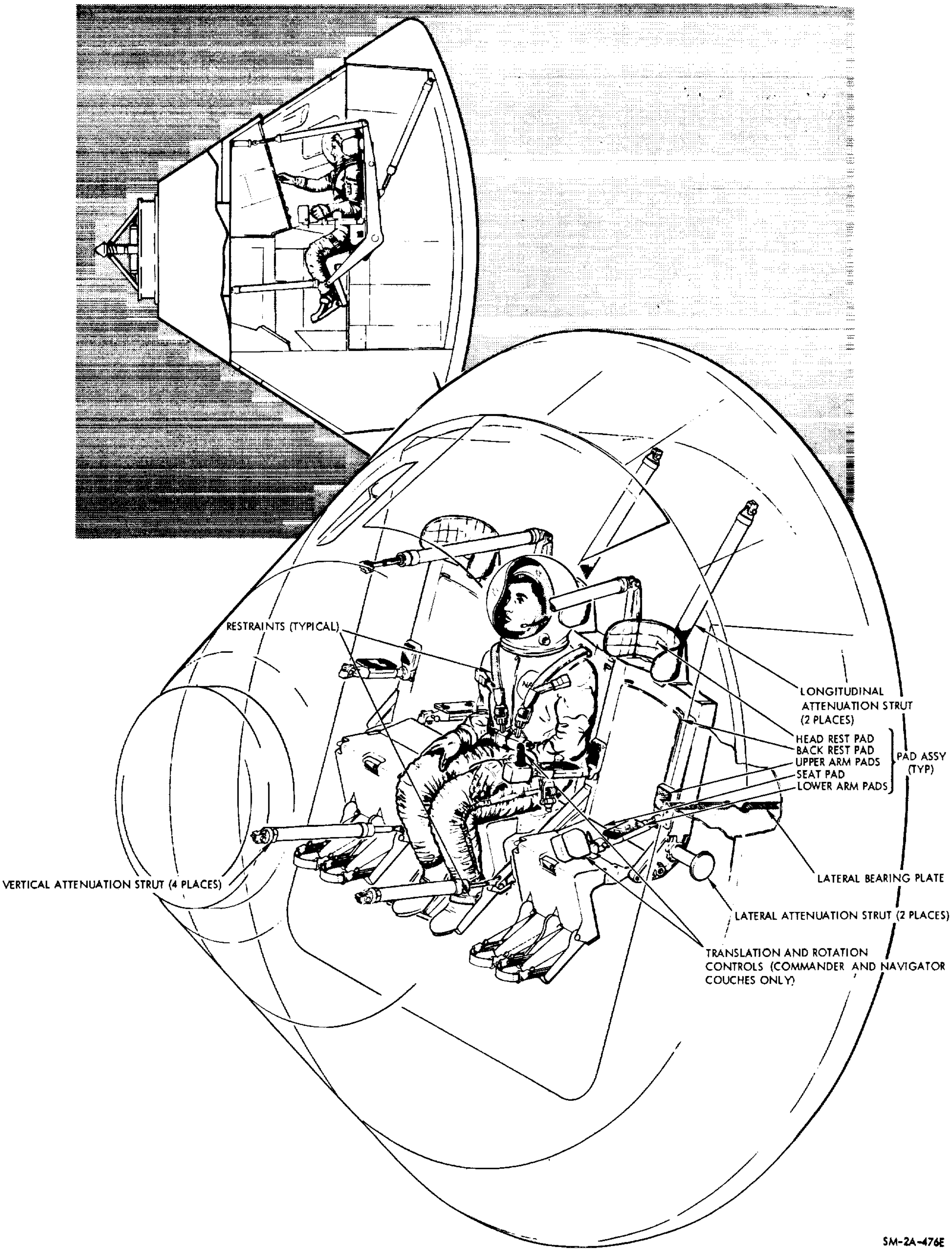


Figure 3-17. Crew Couches and Restraint Equipment

3-87. The bioinstrument accessories kit consists of biomedical sensors, personal and applied, and biomedical preamplifiers. Purpose of the sensors is to acquire the electrical signal required to determine the respiration rate and electrocardiograms of an astronaut. The preamplifier is used to condition and relay the signals generated by the electrocardiograph sensors to the telemetry system for transmittal to earth.

3-88. The radiation dosimeters measure and record the amount of radiation to which the astronaut is exposed. One is located at the right temple of the constant-wear communications assembly; others are located in pockets provided on constant-wear garments.

3-89. The emergency medical kit provides the equipment and medications required for emergency treatment of illness or injuries sustained by crewmen during a mission.

3-90. The physiological clinical monitoring instrument set consists of a sphygmomanometer, a stethoscope, and a thermometer, and is used to measure blood pressure, respiration rate or heart beat, and body temperature.

3-91. The portable life support system is a small self-contained environmental control system and communications unit, and provides life support during extravehicular activities and lunar surface exploration. The PLSS is worn as a backpack and is capable of maintaining the PGA in a pressurized condition for 4 hours without recharging.

3-92. CREW COUCH AND RESTRAINT EQUIPMENT.

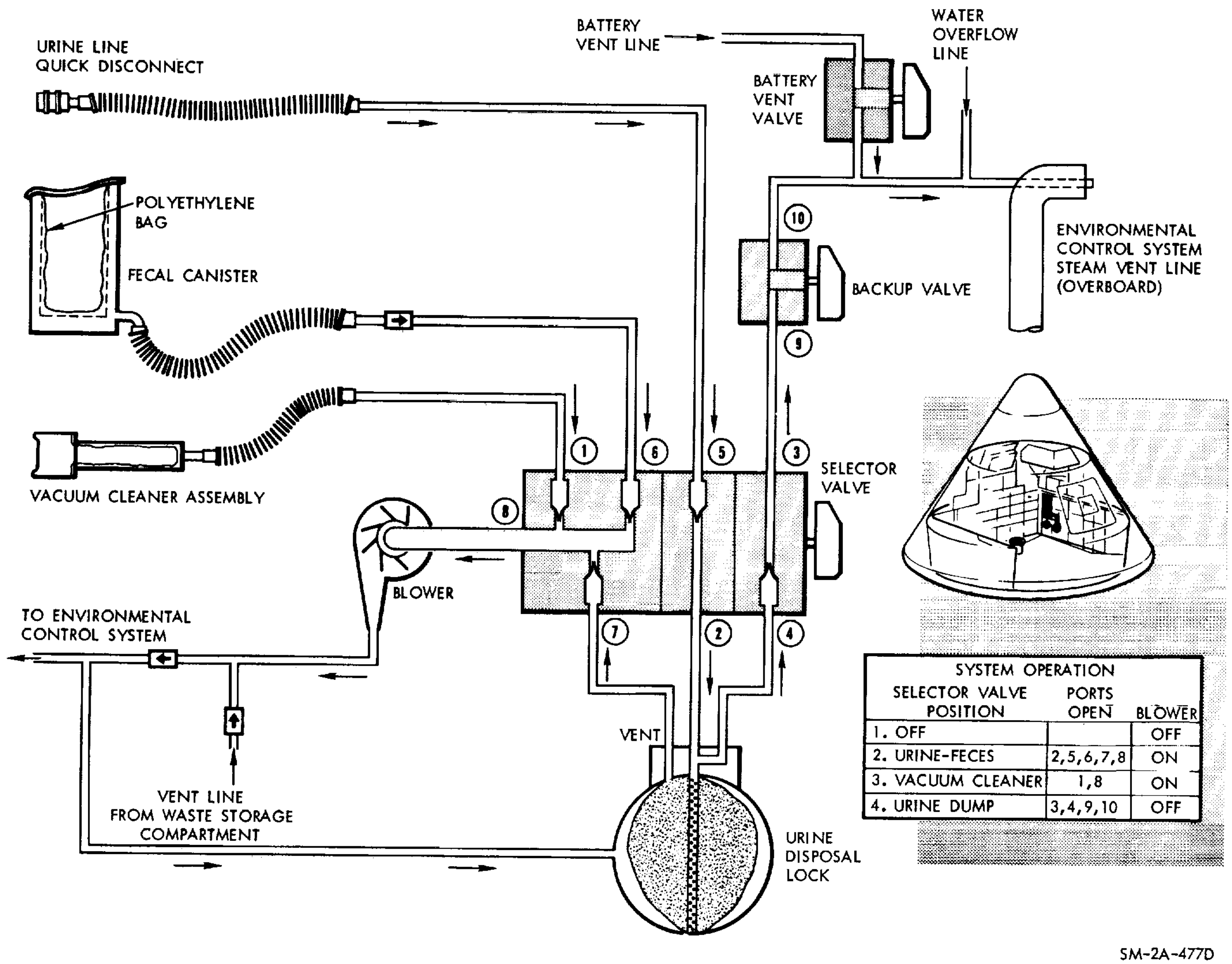
3-93. Crew couch and restraint equipment consists of crew couch pad assemblies, restraint harness assemblies, foot-strap restraint assemblies, and restraint sandals. A pad assembly is installed on each crew couch for crewman comfort. Restraint harness assemblies and foot-strap restraint assemblies are installed on each crew couch primarily to restrain the crewmen during all phases of the mission. Restraint sandals, worn over the feet of constant-wear garments, have soles made of Velcro hook material, which adheres to Velcro pile material installed on the C/M floor.

3-94. WASTE MANAGEMENT SYSTEM.

3-95. The waste management segment of the crew system consists of the means for urine disposal, collecting and storing fecal matter and personal hygienic wastes. Fecal matter is collected in plastic (polyethylene) bags, disinfected, and stored in a vented area. Personal hygienic wastes are also collected and stored in this manner. The urine is temporarily stored and then expelled overboard by the differential pressure method. This is accomplished by properly positioning two manually controlled waste management valves. (See figure 3-18.) A vent line interfaces the WMS to the ECS, and provides for the removal of odors originating as a result of waste management functions. The atmosphere containing these odors is routed through the CO_2-odor absorber filters for processing and re-use within the suit circuit of the ECS.

3-96. CREW SURVIVAL EQUIPMENT.

3-97. Three survival kits are stowed in the C/M and are available to the crew during the postlanding phase (water or land) of a mission. The major items provided in each kit include a container with 6 pounds of water, a desalter kit, a one-man liferaft (one three-man liferaft is provided in Block II C/Ms), a radio-beacon, portable light, sunglasses, machete (and sheath), and a medical kit. The liferaft includes additional equipment such as a sea anchor, dye marker, sunbonnet, etc.



SYSTEM OPERATION		
SELECTOR VALVE POSITION	PORTS OPEN	BLOWER
1. OFF		OFF
2. URINE-FECES	2,5,6,7,8	ON
3. VACUUM CLEANER	1,8	ON
4. URINE DUMP	3,4,9,10	OFF

Figure 3-18. Waste Management System Functional Diagram

3-98. FOOD, WATER, AND ASSOCIATED EQUIPMENT.

3-99. Adequate food, water, and personal hygiene aids will be provided for the total length of the mission. Small polyethylene bags containing frozen and/or dehydrated food will be stored in the C/M. By adding water and kneading, the food mixtures can be squeezed into the crewmember's mouth. Either hot or cold water is available at the potable water supply panel for food reconstitution. Chilled drinking water will be supplied to the astronauts by a single flexible hose assembly from the water delivery unit. This water source, a by-product of the fuel cell powerplants, will furnish the crew up to 36 pounds (17 quarts) of water per day. A folding shelf is provided as a convenient surface for tools, food packages, and equipment. Personal hygiene aids consist of items for oral hygiene and body cleansing such as chewing gum, interdental stimulators, and cleansing pads.

3-100. CREW ACCESSORIES.

3-101. Several accessory items are considered to be part of the crew system. These consist of the following: in-flight tool set, tether line assembly, mirror assemblies for increased interior and exterior vision, main display console handhold straps, and an optical alignment sight. The handhold straps are installed as an aid to crewman mobility within the C/M, and the optical alignment sight is utilized to properly orient and align the CSM with the LEM while accomplishing docking maneuvers.

3-102. COMMAND MODULE INTERIOR LIGHTING.

3-103. Block I C/M interior lighting equipment (figure 3-19) provides light for the main control panels in the command module and consists of eight floodlight fixture assemblies and three control panels. Each fixture assembly contains two fluorescent lamps (one primary and one secondary) and a converter. The interior lighting is powered by d-c main buses A and B, assuring a power source for lights in all areas in the event that either bus fails. The converter in each floodlight fixture converts 28 volts dc to a-c power to operate the fluorescent lamps. The floodlights are used to light three areas: the main display console (left and right areas) and the LEB area.

3-104. Each control panel has a primary and secondary control for the floodlights in its respective area. The primary control is a rheostat that controls brightness of the primary floodlights. The secondary control is an on-off switch for the secondary floodlights and is set to on when additional brightness is desired.

3-105. Interior lighting for Block II S/C (figure 3-19) is essentially the same as Block I S/C, with the addition of electroluminescent lighting to the control and display panels, and the forward tunnel. The electroluminescent lights are used for low-level floodlighting, and the fluorescent lights are used for high-level floodlighting.

3-106. TELECOMMUNICATION SYSTEM.

3-107. The function of the telecommunication system (figure 3-20) is to provide for the communication of voice, television, telemetry, and tracking and ranging data between the S/C and the MSFN, the LEM and EVA PLSS. It also provides for S/C intercommunications and, includes the central timing equipment for synchronization of other equipment and correlation of telemetry data. The T/C system contains the following equipment, listed in four groups:

a. Data equipment group

- Signal conditioning equipment (SCE)
- Pulse code modulation/telemetry (PCM/TLM) equipment
- Television (TV) equipment
- Up-data link (UDL) equipment
- Premodulation processor (PMP) equipment
- Data storage equipment (DSE)
- Flight qualification recorder (FQR)
- Central timing equipment (CTE)

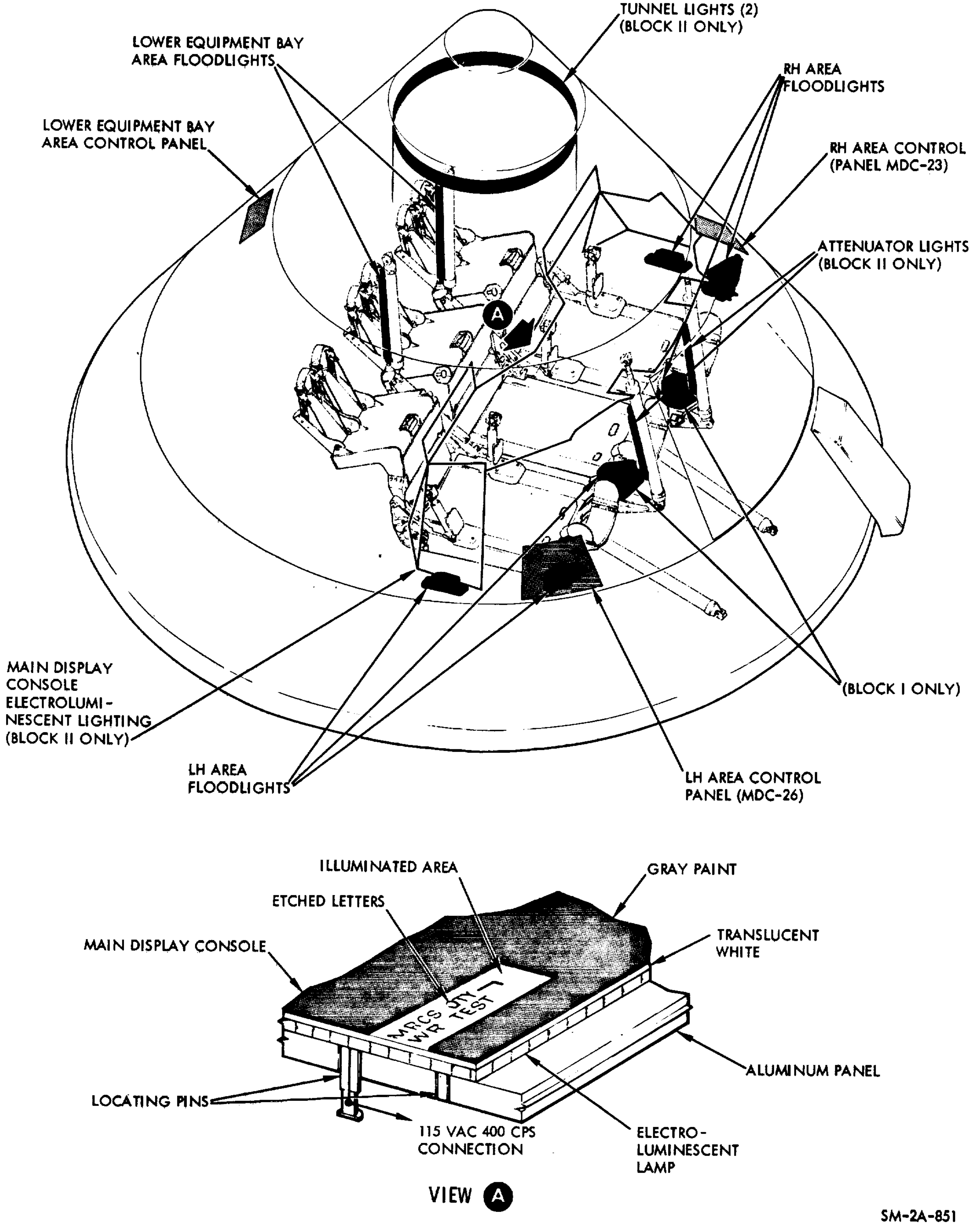


Figure 3-19. Command Module Interior Lighting Configuration

b. Intercommunications equipment group

- Audio center equipment
- Headsets and connecting electrical umbilicals

c. RF electronics equipment group

- VHF/AM transmitter - receiver equipment
- VHF/FM transmitter equipment
- HF transceiver equipment
- VHF recovery beacon equipment
- Unified S-band equipment (USBE)
- S-band power amplifier (S-band PA) equipment
- C-band transponder equipment (Block I only)
- Rendezvous radar transponder equipment (Block II only)

d. Antenna equipment group

- VHF/2-KMC omni-antenna equipment
- 2-KMC high-gain antenna equipment (Block II only)
- VHF recovery antenna equipment
- HF recovery antenna equipment
- C-band beacon antenna equipment (Block I only)
- Rendezvous radar antenna equipment (Block II only)
- Ancillary equipment

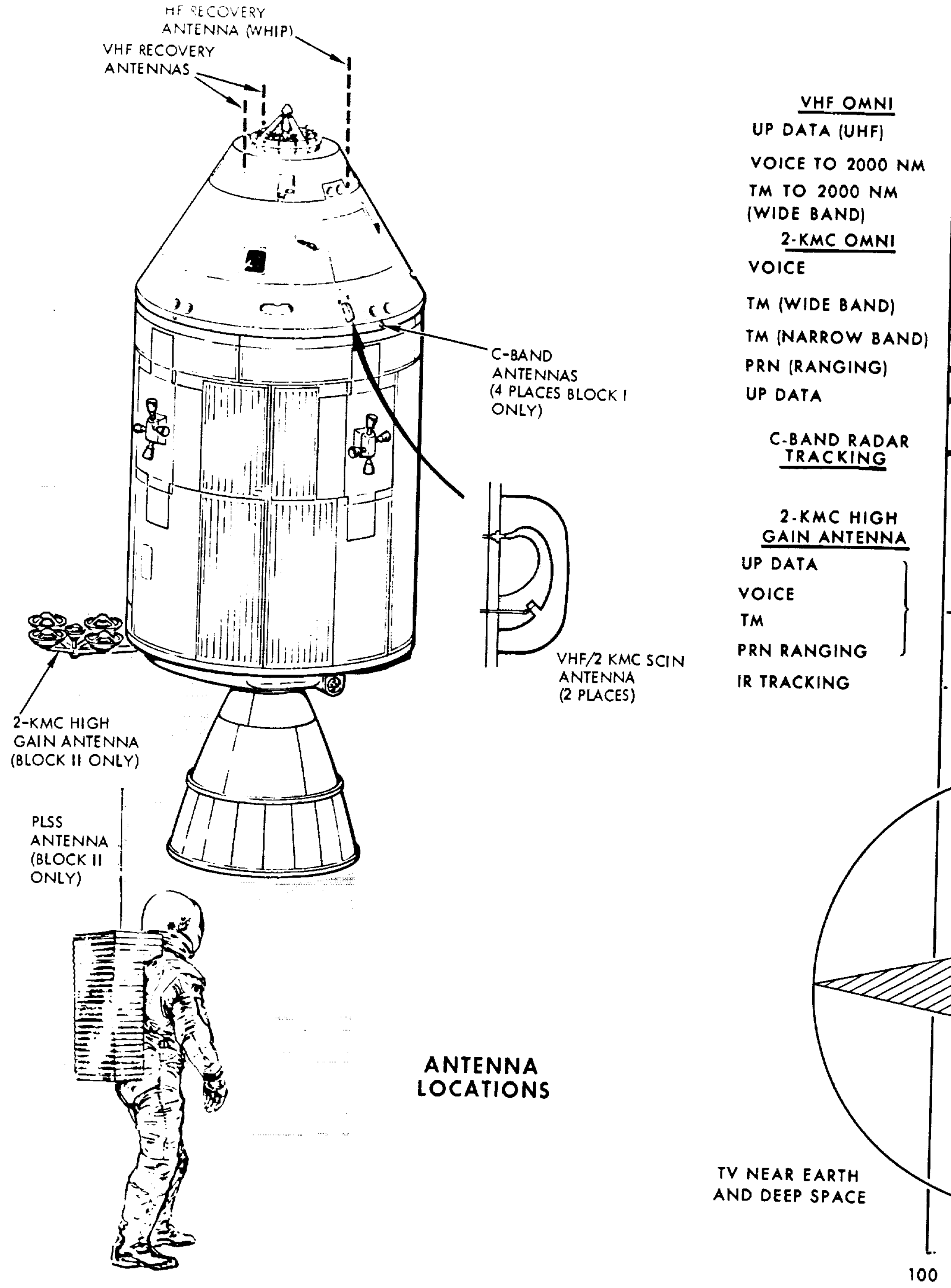
Controls and switches for operation of the T/C system are located near the engineer's station in the crew compartment. Also, there are three separate groups of controls (one for each crewmember) for individual control of audio inputs and outputs of the crewmembers headsets.

3-108. VOICE OPERATIONS.

3-109. S/C voice communications originate and terminate in the astronauts headsets. They are used for all voice transmission and reception, S/C intercommunications, and hardline communications with the launch control center (LCC) during prelaunch checkout. Each astronaut has an individual audio control panel, on the main display console (MDC), which enables him to select and control the inputs and outputs of his headset.

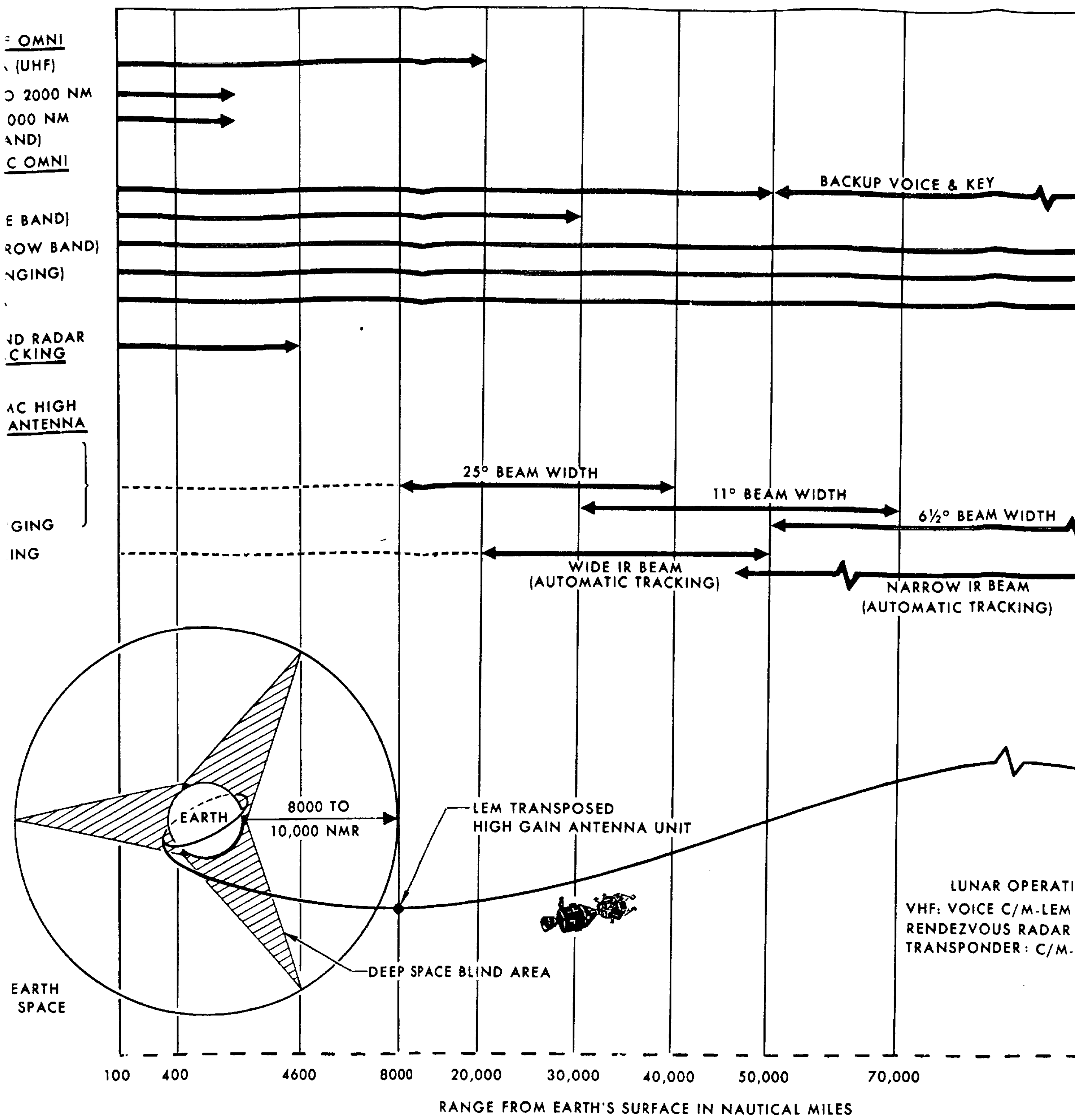
3-110. The headsets and audio control panels are connected to the audio center equipment which contains three identical audio center modules, one for each audio control panel and headset. The audio center equipment serves as the common assimilation and distribution point for all S/C audio signals. It is controlled by the audio control panels and the T/C controls on MDC panel No. 20. Depending on the mode of operation, the audio signals are routed to and from the applicable transmitter and receiver, the LCC, the recovery forces intercom, or the data storage equipment.

3-111. Three methods of voice transmission and reception are possible: the VHF/AM transmitter-receiver, the HF transceiver, or the S-band transmitter and receiver in the unified S-band equipment. Transmission is controlled by either the push-to-talk (PTT) switch, located in the electrical umbilical cord provided for each astronaut, or the voice-operated relay (VOX) circuitry. The PTT switch also serves as a keying switch during USBE emergency key transmission.



ANTENNA LOCATIONS

COMMUNICATION RANGES



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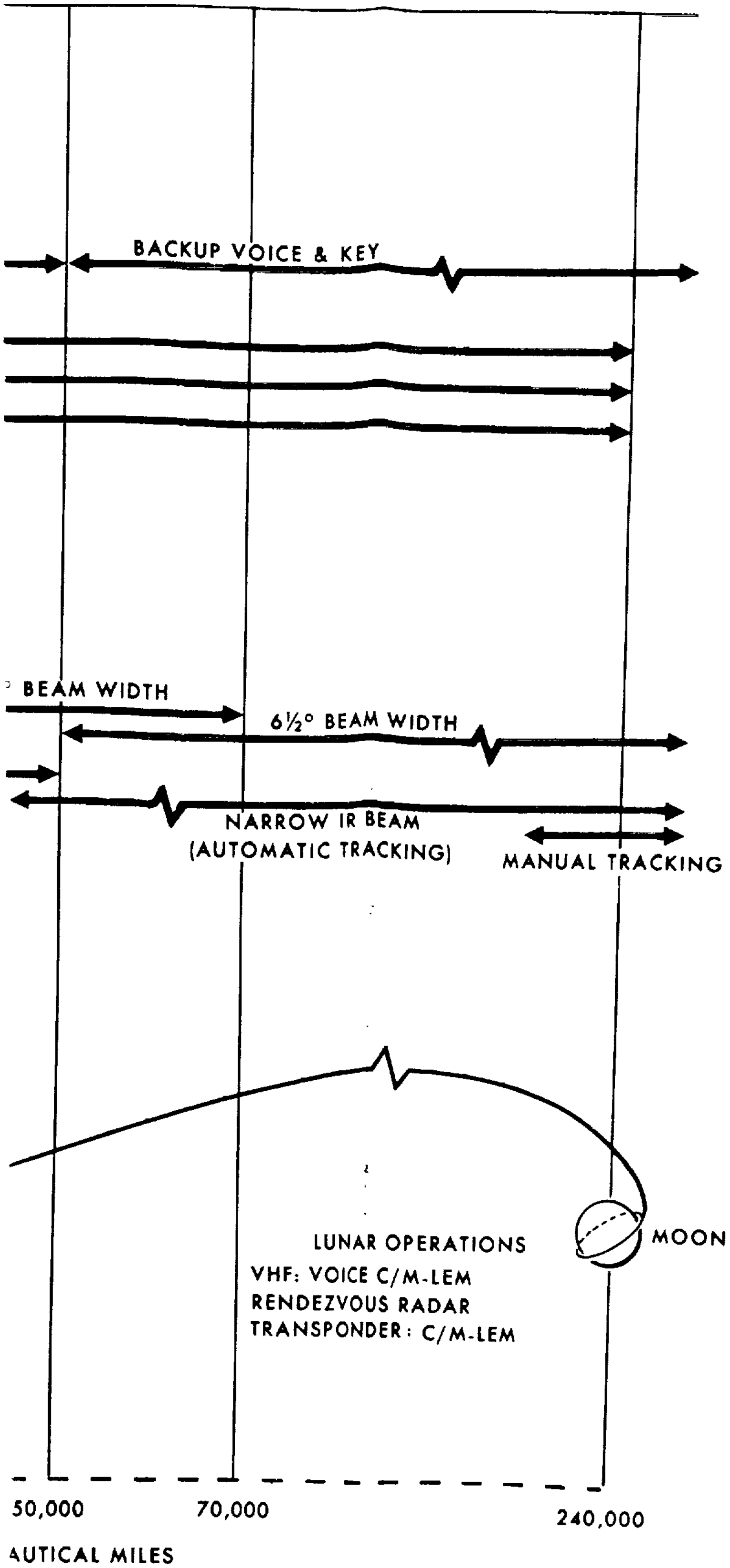

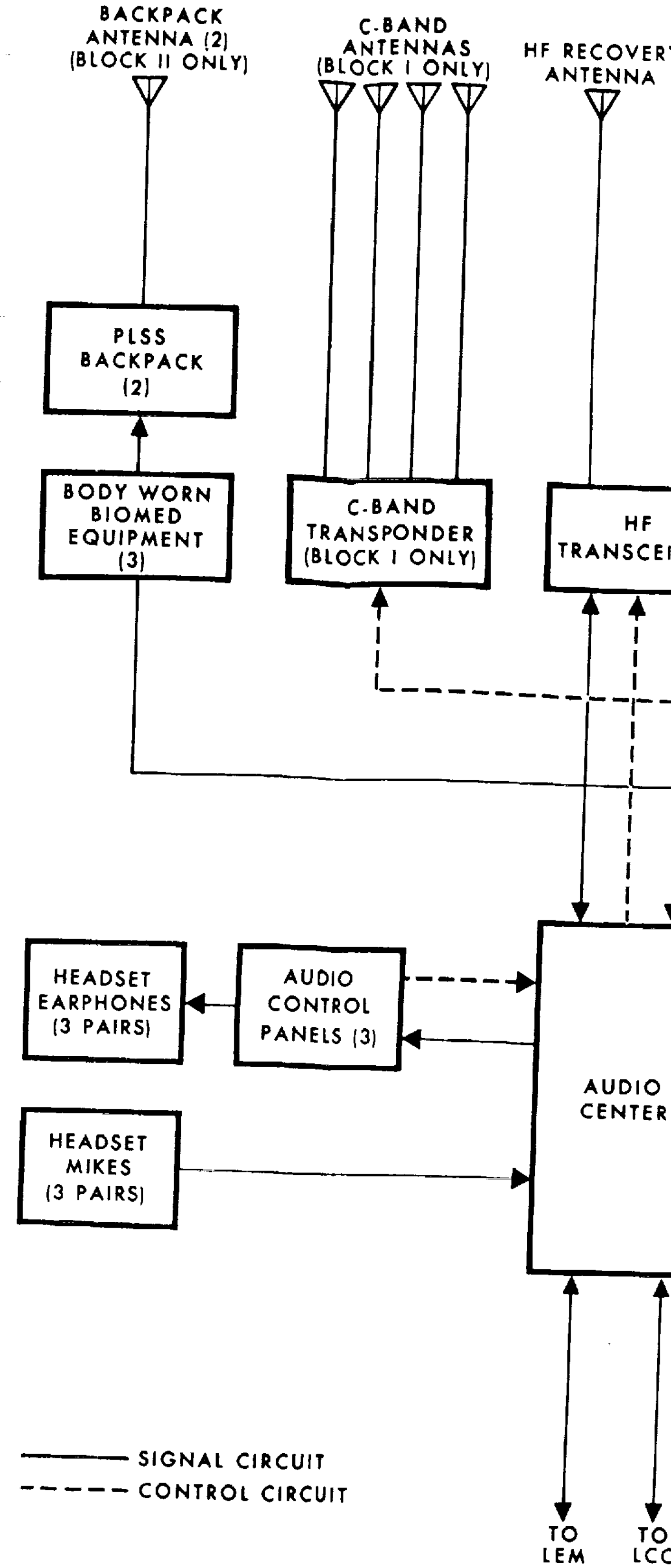

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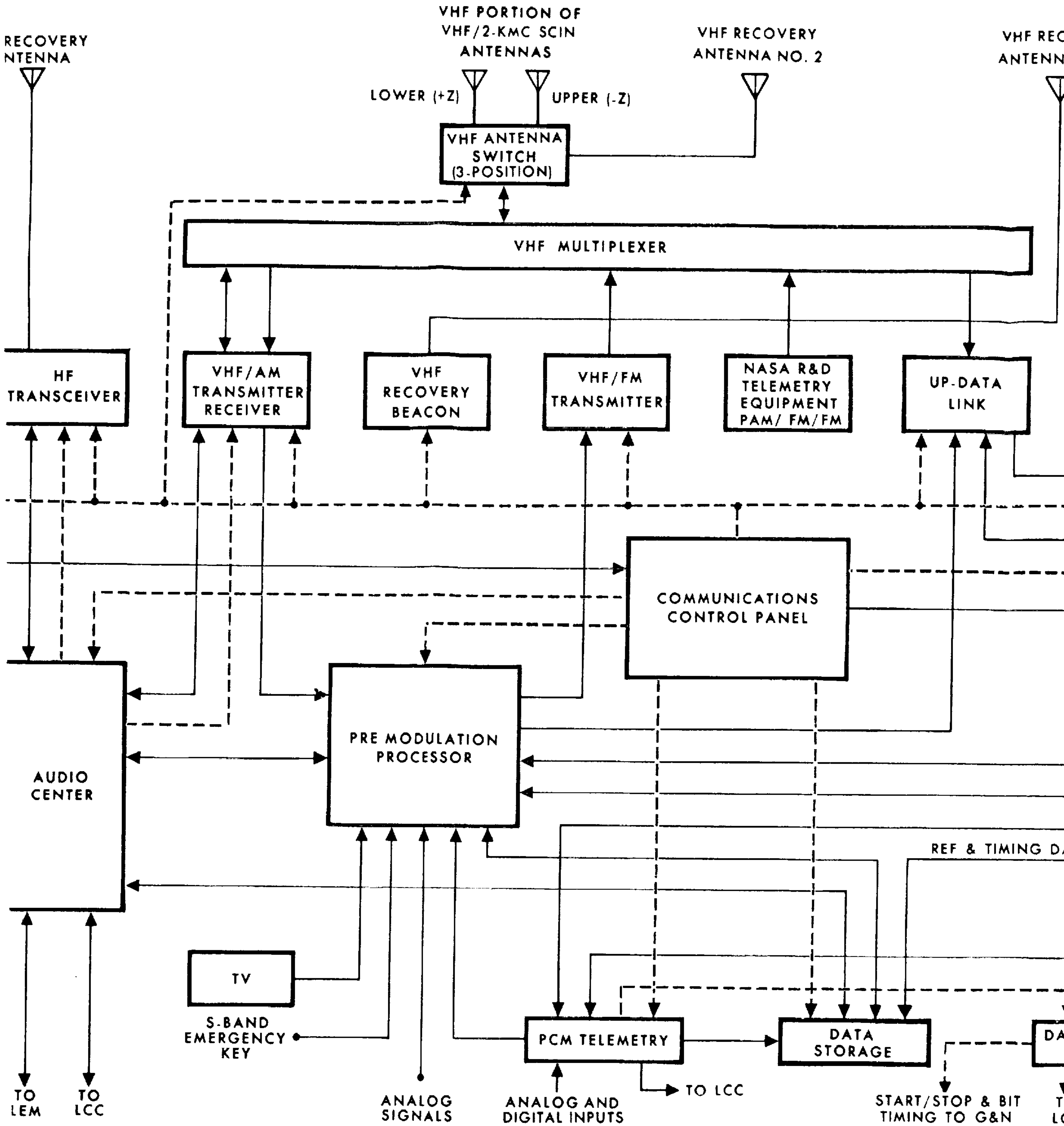

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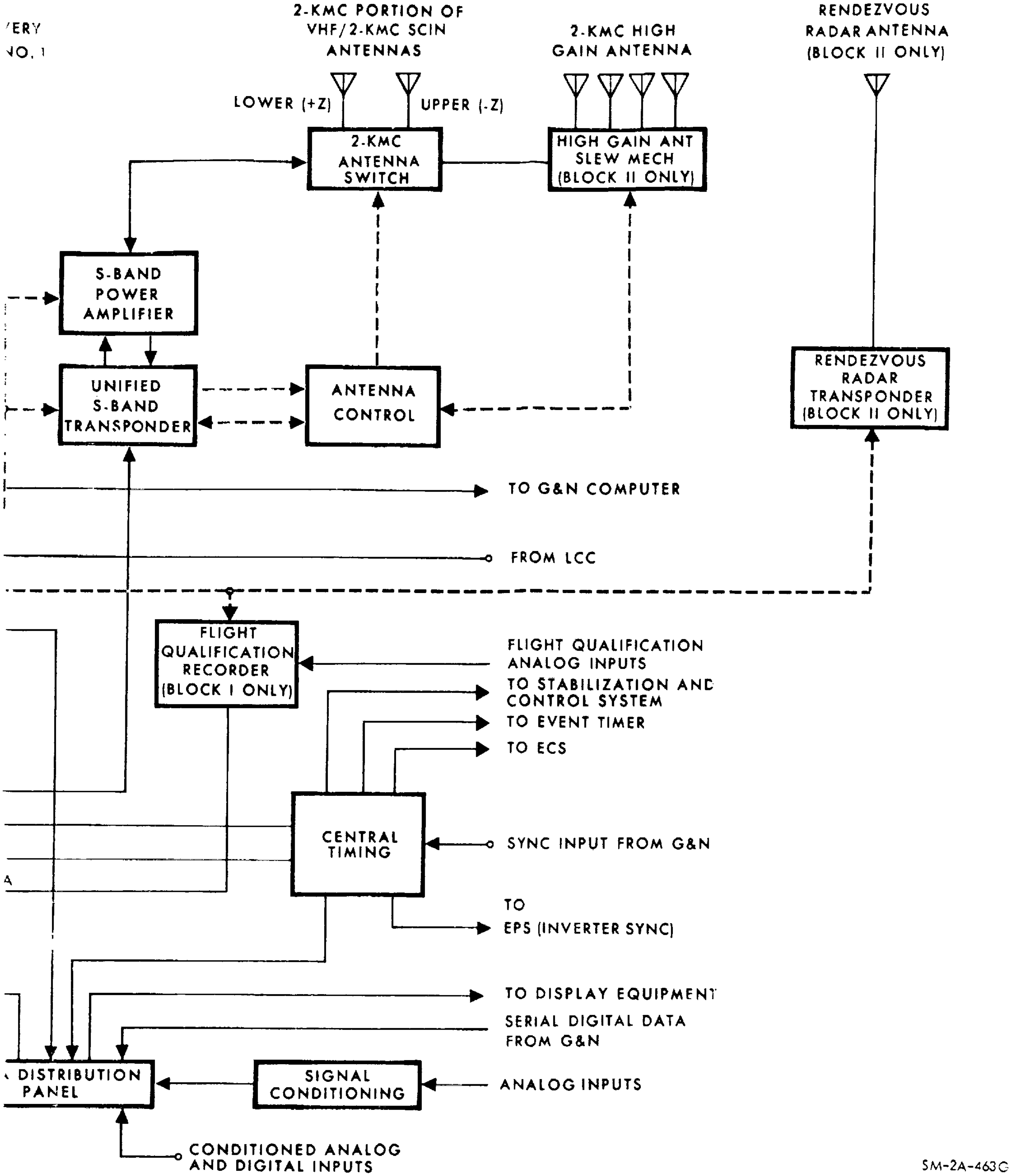


Figure 3-20. Telecommunications Systems - Antenna Location, Range, and Block Diagram

3-112. The VHF/AM transmitter-receiver equipment is used for voice communications with the MSFN during launch, ascent, and near-earth phases of the mission. The USBE is used during deep space phases of the mission when the S/C is not within VHF range of a ground station. When communications with the MSFN are not possible, limited capability exists to store audio signals on tape in the DSE for later transmission or playback on the ground after the mission is completed. For recovery operations during the post-landing phase of the mission, voice communications with the MSFN and recovery personnel are conducted over the VHF/AM transmitter-receiver equipment, the HF transceiver equipment, or the recovery forces intercom via the swimmers umbilical connector in the C/M forward compartment.

3-113. DATA OPERATIONS.

3-114. The S/C structure and operational systems are instrumented with sensors and transducers which gather data on their physical status. Biomedical data from sensors worn by the astronauts, TV data from the TV camera, and timing data from the central timing equipment are also acquired. These various forms of raw data are assimilated into the system, processed, and then transmitted to the MSFN. Data from the operational instrumentation may be stored in the DSE for later transmission or analysis. Analog data from the flight qualification instrumentation is stored in the flight qualification recorder for postflight analysis only.

3-115. Unconditioned analog and on-off event signals from instrumentation sensors are fed into the signal conditioning equipment where they are conditioned to a standard 5-volt d-c level. These signals are then sent to the data distribution panel which routes them to the pulse code modulation telemetry equipment and C/M displays. The PCM telemetry equipment combines the inputs from the SCE with other low-level analog inputs and converts them to a single, digital, modulating signal which is then routed to the premodulation processor equipment.

3-116. The PMP is the common assimilation, integration, and distribution center for nearly all forms of S/C data and provides the necessary interface with the RF equipment. In addition to the input from the PCM telemetry equipment, the PMP accepts recorded PCM and analog data from the DSE, video signals from the TV equipment, timing signals from the CTE, and audio signals from the audio center equipment when audio is transmitted by the USBE. These signals are modulated, mixed, and switched to the applicable transmitter or the DSE according to the mode of operation. Voice signals and up-data commands received over the USBE receiver from the MSFN are also supplied to the PMP which routes them to the audio center equipment and up-data link equipment, respectively. The UDL also contains its own receiver which is normally used during near-earth phases of the mission. S-band reception of up-data commands is used in deep space.

3-117. PCM/TLM data is transmitted to the MSFN by the VHF/FM transmitter equipment during near-earth phases of the mission. Transmission of TV or analog data is possible only by the USBE which is normally used in deep space but can also be used during launch and near-earth phases of the mission when within sight of an S-band equipped ground station. If the USBE is used, PCM/TLM data and voice signals can be transmitted with TV or analog data over the S-band link.

3-118. TRACKING AND RANGING OPERATIONS.

3-119. The T/C tracking and ranging equipment assists the MSFN in accurately determining the angular position and range of the S/C. Two methods are used: C-band tracking and S-band tracking.

3-120. C-band tracking is accomplished by the C-band transponder equipment and is used during all near-earth phases of the mission. It operates in conjunction with earth-based radar equipment. The C-band transponder transmits an amplified RF pulse in response to a properly coded, pulsed interrogation from the radar equipment. The range and accuracy of this equipment is thereby greatly extended over what would be possible by using skin-tracking techniques.

3-121. S-band tracking utilizing the USBE transponder is used in deep space. It operates in conjunction with MSFN equipment by providing responses to properly coded interrogations from the earth. When the USBE is in a ranging mode, the USBE transponder will receive PRN ranging code signals from the MSFN and respond by transmitting a similar signal. This method is used initially to establish an accurate measure of range to the S/C and subsequently at periodic intervals to update the doppler ranging data at the MSFN. The doppler measurements are obtained from the S-band carrier.

3-122. A VHF recovery beacon and an HF transceiver is provided to aid in locating the S/C during the recovery phase of the mission. The VHF recovery beacon equipment provides for line-of-sight direction finding capabilities by emitting a 2-second, modulated, RF transmission every 5 seconds. The HF transceiver equipment can be operated in a beacon mode to provide for beyond-line-of-sight direction finding by emitting a continuous wave signal.

3-123. INSTRUMENTATION SYSTEM.

3-124. The instrumentation system consists of those means required for the collection of data and is comprised of: operational, special, flight qualification, and scientific instrumentation. Equipment requirements include a variety of sensors, transducers, and photographic equipment which will be qualified prior to manned flight. Sensors and transducers are used for converting physical and electrical measurements into electrical signals. These signals are conditioned (by signal conditioners) to proper values for distribution to data utilization equipment and S/C displays.

3-125. Sensors and transducers, located strategically throughout the S/C, are positioned on the structure, within the operational systems, and for biomedical purposes are attached to the astronauts. Data may be transmitted to MSFN by way of the telecommunications system, displayed to the astronauts, or stored for evaluation at the completion of the mission.

3-126. The photographic equipment carried aboard the command module is provided for still photo and moving picture coverage.

3-127. OPERATIONAL INSTRUMENTATION.

3-128. Operational instrumentation consists of approximately 24 classes of transducers and is based on the following specific measurements:

Pressure	Attitude	Angular Position	Voltage
Temperature	Rates	Current	Frequency
Flow	Events	Quantity	RF Power

3-129. SPECIAL INSTRUMENTATION.

3-130. Special instrumentation consists of the equipment required for the checkout and monitoring of proton radiation detection and gas chromatograph. Requirements for additional equipment such as photographic, biomedical, fire-detection, and anthropomorphic dummies will vary, depending on the mission.

3-131. SCIENTIFIC INSTRUMENTATION.

3-132. Scientific instrumentation consists of the equipment required for various scientific experiments. The photographic equipment included in this group consists of a 35-mm still-photo camera and a 16-mm movie camera. Also included are data recorders which are installed when required.

3-133. FLIGHT QUALIFICATION.

3-134. Flight qualification consists of evaluation tests to ensure that the systems will furnish the instrumentation system with the required information in the specified form, from the source to the utilization point. The systems to be tested include sensors, associated equipment, subsystems, and special instrumentation such as optical, scientific, and biomedical. This data will be recorded on the flight qualification recorder for post-flight analysis.

3-135. CAUTION AND WARNING SYSTEM.

3-136. The caution and warning system (C&WS) monitors critical parameters of most S/C systems. Each malfunction or out-of-tolerance condition is brought to the attention of the crew by visual and aural means. The crew acknowledges the condition and resets the system for subsequent malfunction alerting.

3-137. C&WS OPERATION.

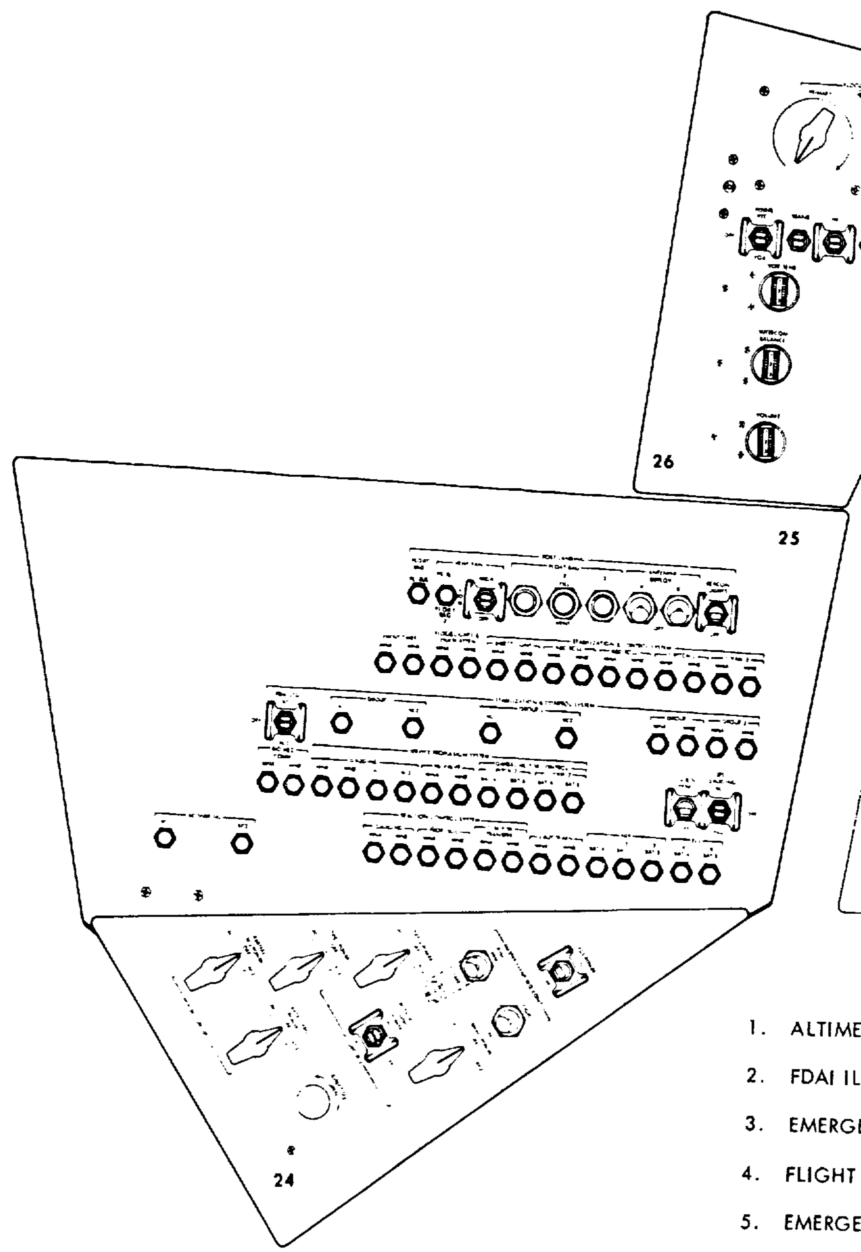
3-138. Malfunctions or out-of-tolerance conditions are sent to the C&WS as analog and discrete event inputs. This results in illumination of status lights that identify the abnormal condition, and activation of the master alarm circuit. Master alarm lights on the panels and an audio alarm tone in the headsets serve to alert the crew to each abnormal condition. Crew acknowledgement of the condition consists of resetting the master alarm circuit, thereby placing it in readiness should other malfunctions occur. C&WS operational modes are selected by the crew to meet varying conditions during the mission. The system also contains its own failure sensing signal.

Figure 3-21. Controls and Displays - Main Display Console (Sheet 1 of 3)

3-139. CONTROLS AND DISPLAYS

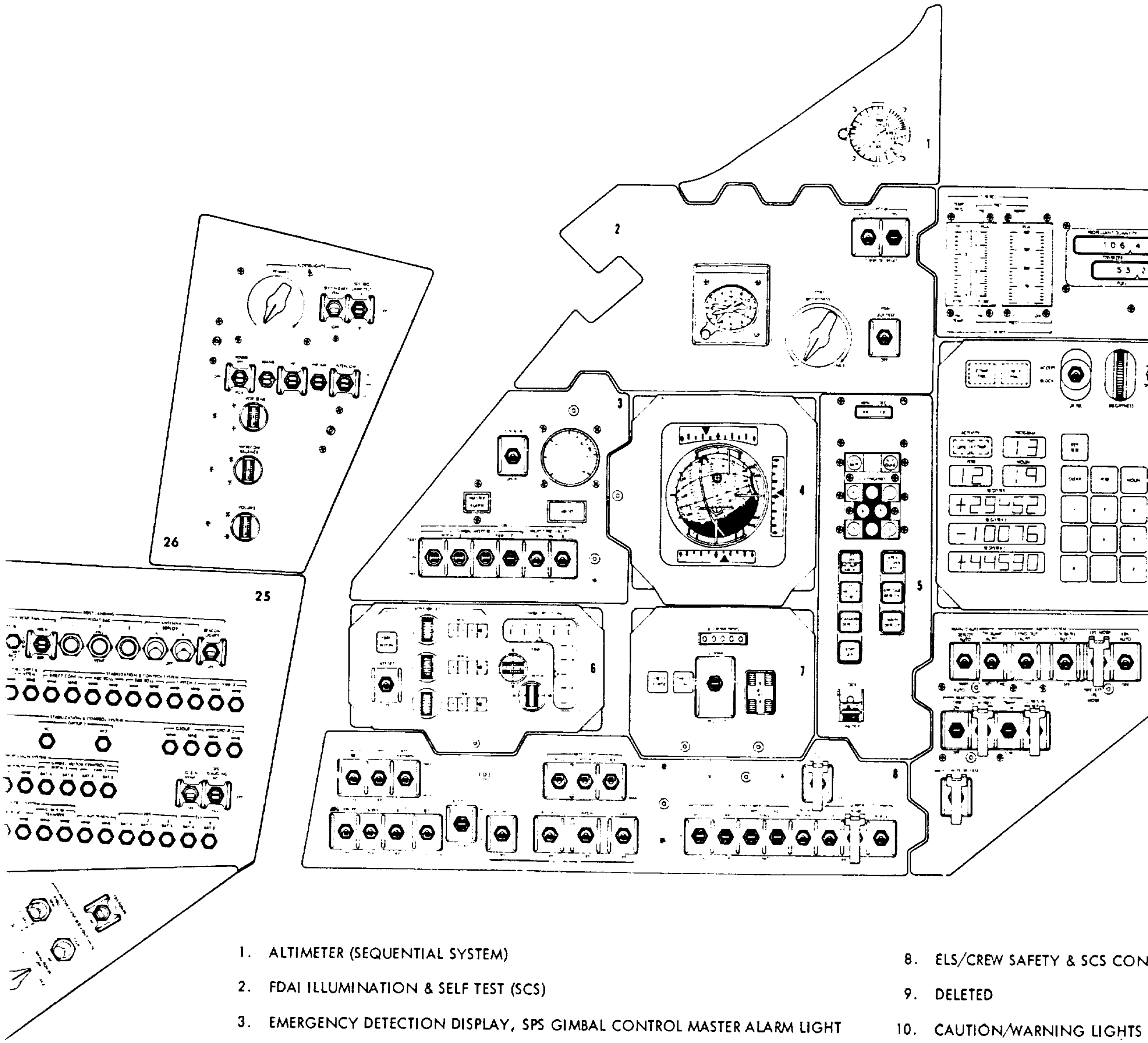
3-140. The operational spacecraft systems, located in the C/M and in the S/M, interface in varying degrees with control and display panels in the C/M cabin. This interface is provided to the extent necessary for the astronauts to adequately control and monitor the functions of the various systems. The controls and displays for most of the S/C systems are located on the main display console situated above the couches. (See figure 3-21.) This location permits frequent attention and quick control by the astronauts. Several of the S/C systems also have additional controls and displays elsewhere in the C/M cabin, as shown in figure 3-22. The majority of guidance and navigation system controls and displays are located on panels in the lower equipment bay adjacent to the G&N telescope and sextant. Those manual controls of the environmental control system that do not require frequent or time-critical actuation, are located in the left-hand equipment bay and the left-hand forward equipment bay. All of the controls in the waste management segment of the crew system are located on a panel in the right-hand equipment bay.

3-141. The displays that read out within given parameters are range-marked to aid the crew in more rapidly determining slight out-of-tolerance conditions. This is in addition to the caution/warning lights that will call the crew's attention to out-of-tolerance conditions. Regardless of the various types of controls in the C/M, provisions have been made for all controls to be operated with an astronaut's gloved hand.



1. ALTIME
2. FDAI IL
3. EMERGE
4. FLIGHT
5. EMERGE
6. ATTITUD
7. ΔV DISP

FOLDOUT FRAME 1



1. ALTIMETER (SEQUENTIAL SYSTEM)
2. FDAI ILLUMINATION & SELF TEST (SCS)
3. EMERGENCY DETECTION DISPLAY, SPS GIMBAL CONTROL MASTER ALARM LIGHT
4. FLIGHT DIRECTOR ATTITUDE INDICATOR UNIT
5. EMERGENCY DETECTION & SEQUENCER SYSTEM DISPLAY
6. ATTITUDE SET & GIMBAL POSITION DISPLAY UNIT
7. ΔV DISPLAY UNIT
8. ELS/CREW SAFETY & SCS CON
9. DELETED
10. CAUTION/WARNING LIGHTS
11. CAUTION/WARNING LIGHTS
12. RCS & MISSION ELAPSED TIME
13. CAUTION & WARNING, AUDI
14. G&N COMPUTER CONTROL U

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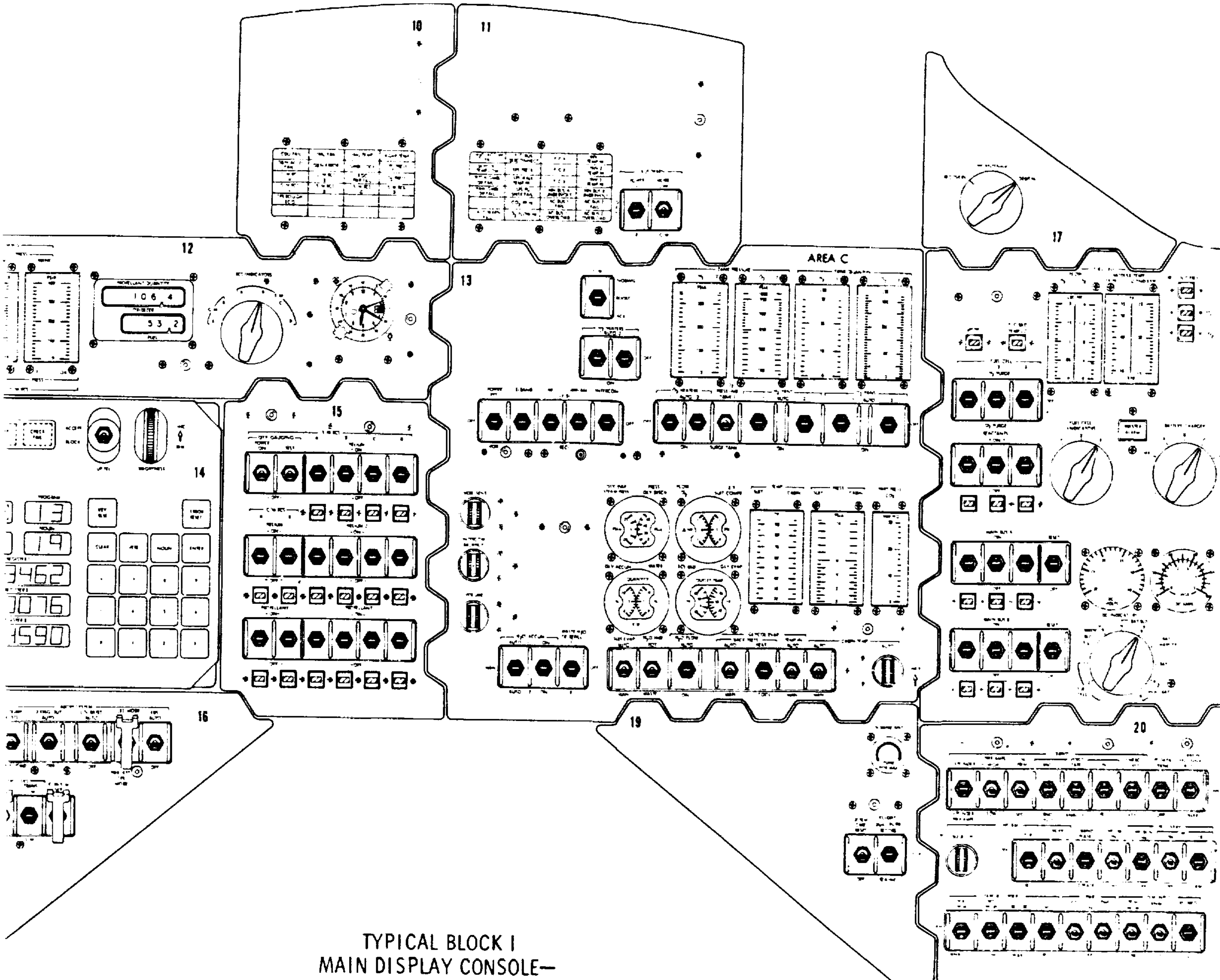

TYPICAL BLOCK I
MAIN DISPLAY CONSOLE—
PANEL IDENTIFICATION LIST

:W SAFETY & SCS CONTROL UNIT

)

N/WARNING LIGHTS

N/WARNING LIGHTS

ISSION ELAPSED TIMER

N & WARNING, AUDIO, CRYOGENIC & ECS

)MPUTER CONTROL UNIT

15. C/M & S/M RCS CONTROL

16. CREW SAFETY CONTROL

17. HF ORBITAL ANTENNA CONTROL

18. FUEL CELLS, EPS & MASTER ALARM LIGHT

19. S-BAND ANTENNA DISPLAY

20. COMMUNICATIONS, DATA STORAGE, S-BAND ANTENNA SELECT & SPS PLUGS

21. RH SIDE CONSOLE - CIRCUIT BREAKER & BUS SWITCHING

22. RH SIDE C

23. RH SIDE C

24. LH SIDE C

25. LH SIDE C

26. LH SIDE C

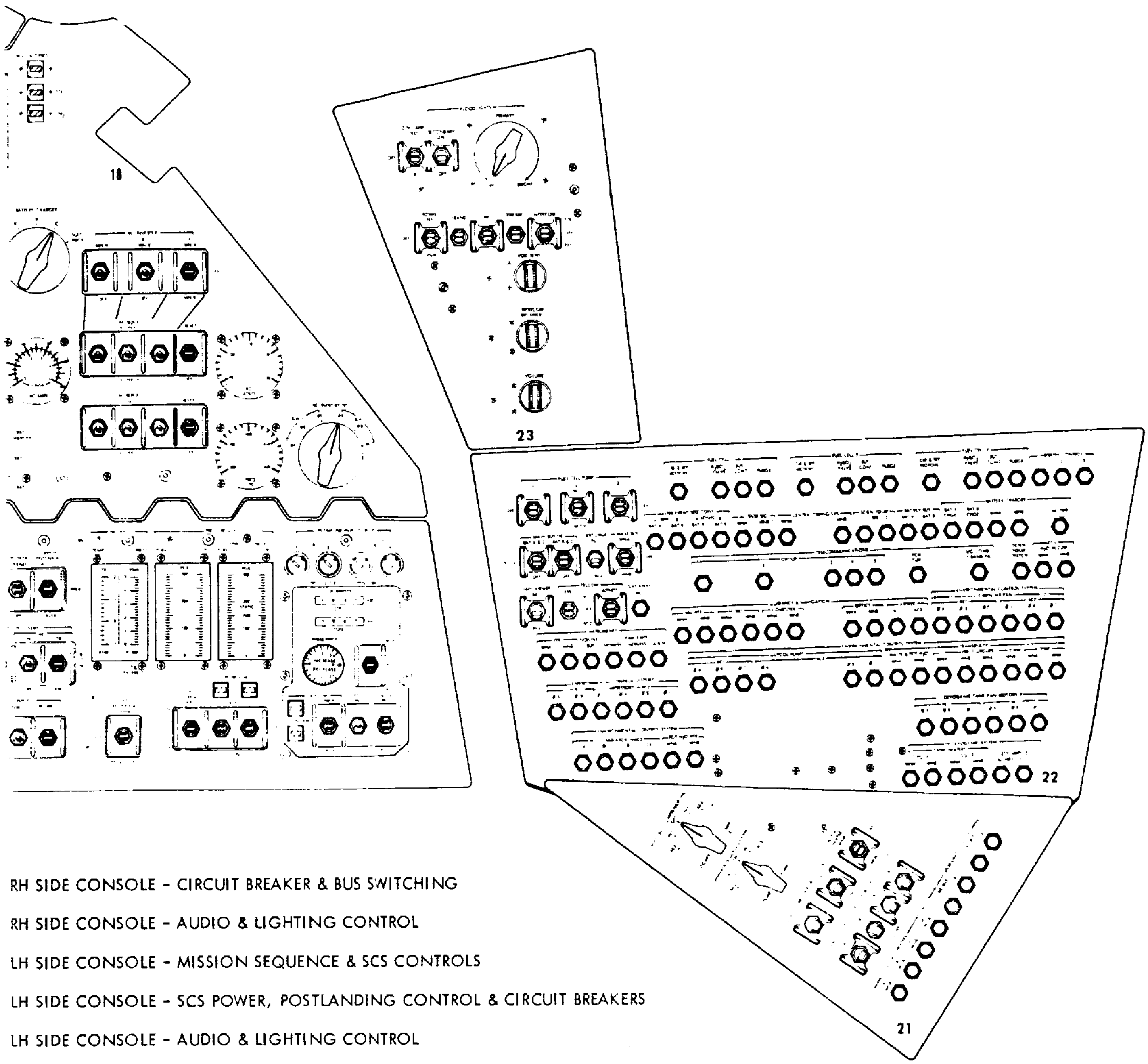

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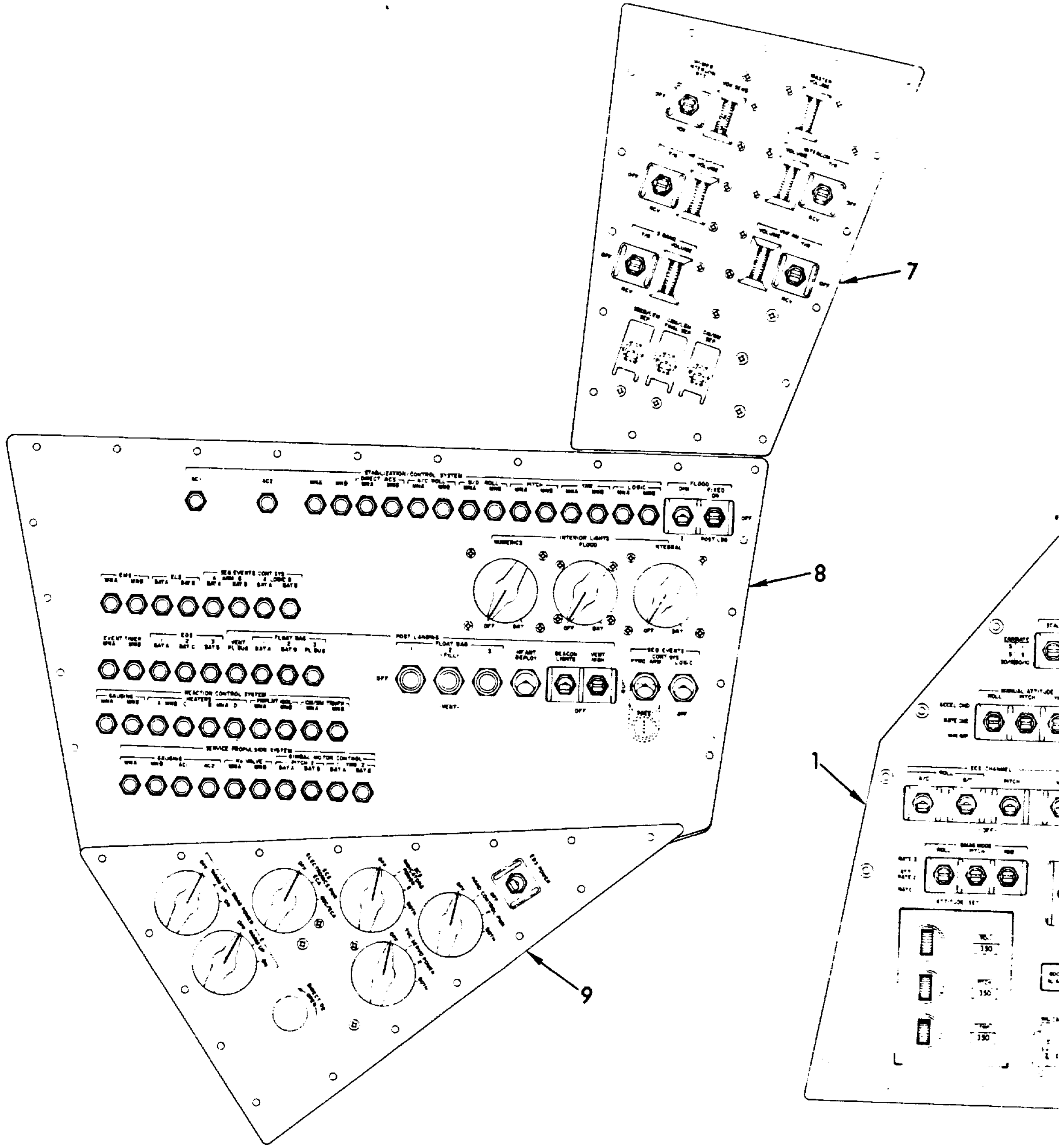
Figure 3-21. Controls and Displays - Main Display Console (Sheet 2 of 3)



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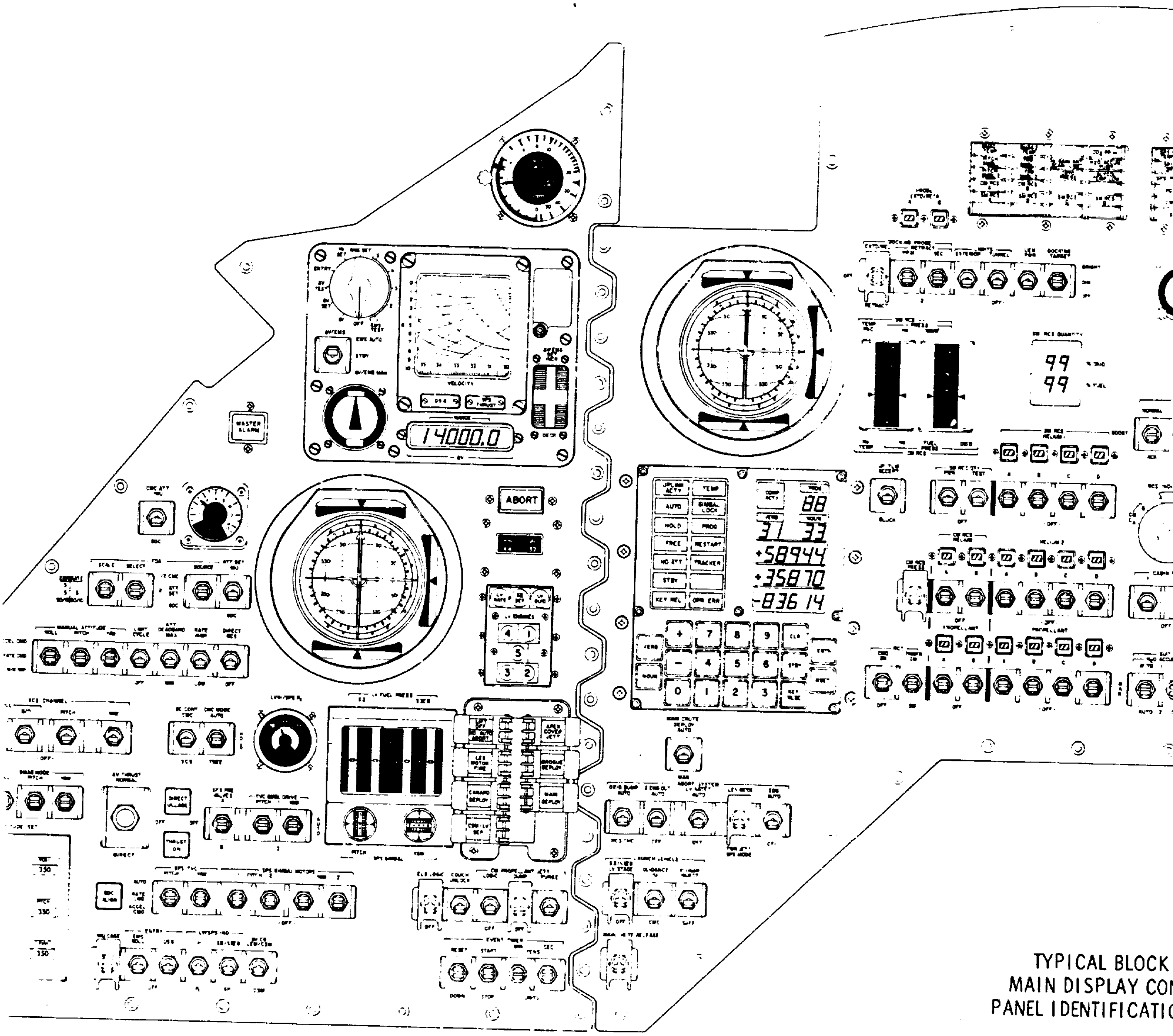


TYPICAL BLOCK
MAIN DISPLAY CO
PANEL IDENTIFICATI

1 ALTIMETER, EMS AND ΔV REMAINING INDICATOR, FDAI, L/V FUEL AND ENGINES, SEQUENCER SYSTEM CONTROLS AND DISPLAYS, SPS GIMBAL CONTROL, ATTITUDE SET, G AND C, MASTER ALARM LIGHT, AND ABORT LIGHT.

2 CAUTION/WARNING LIGHTS, FDAI, G AND C DSKY, SEQUENCER SYSTEM, DOCKING SYSTEM, S/M AND C/M RCS, PHASE AND MISSION ELAPSED TIMERS, CRYOGENIC SYSTEM, ECS, AND HIGH GAIN ANTENNA.

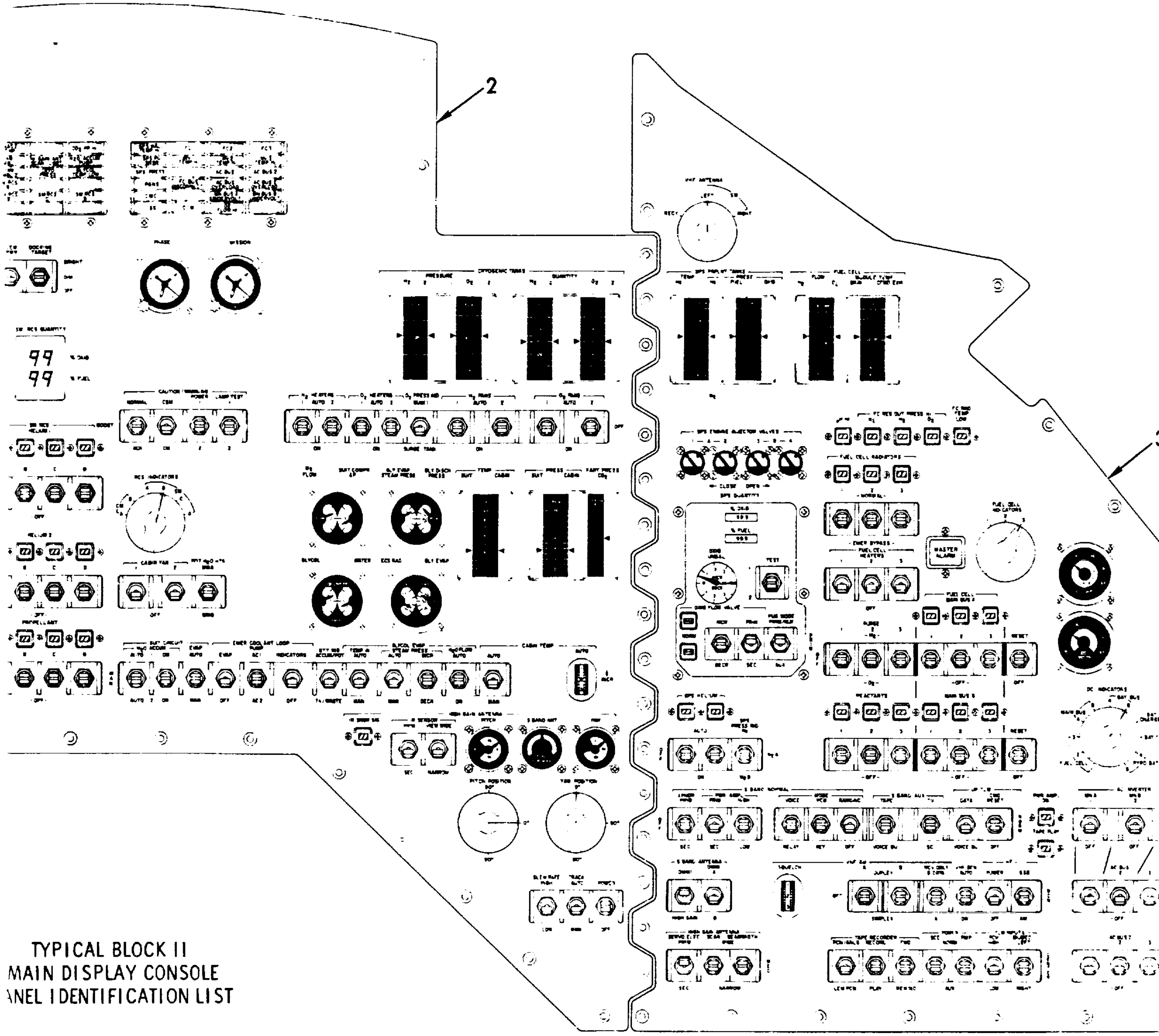
3 SPS, FUEL CELLS, EPS, T/C, AND MASTER

4 RH SIDE CONSOLE-SPS, T/C, AND ECS, B CIRCUIT BREAKERS.

5 RH SIDE CONSOLE-BUS SWITCHING, CIRC INTERIOR LIGHT CONTROLS.

6 RH SIDE CONSOLE-T/C CONTROLS.

FOLDOUT FRAME 2



TYPICAL BLOCK II
MAIN DISPLAY CONSOLE
\NEL IDENTIFICATION LIST

PS, T/C, AND MASTER ALARM LIGHT.

-SPS, T/C, AND ECS, BUS SWITCHING AND ECS

-BUS SWITCHING, CIRCUIT BREAKERS, AND
ͻNTROLS.

-T/C CONTROLS.

7 LH SIDE CONSOLE-EDS AND SCS CONTROLS.

8 LH SIDE CONSOLE-SCS, SEQUENCER SYSTEM, EDS, RCS, AND SPS CIRCUIT BREAKERS, POSTLANDING CIRCUIT BREAKERS AND SWITCHES, AND INTERIOR LIGHT CONTROLS.

9 LH SIDE CONSOLE-T/C CONTROLS.

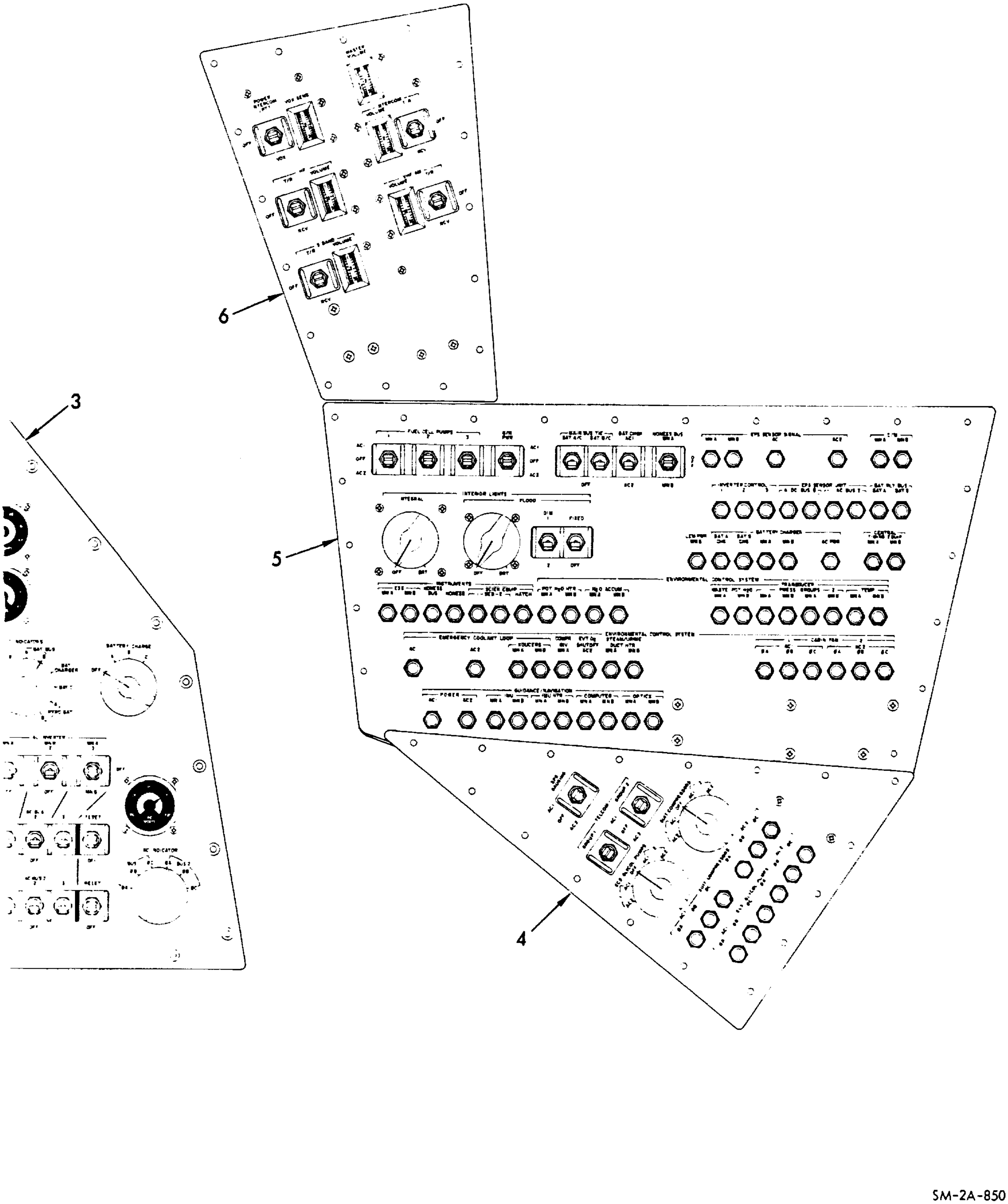
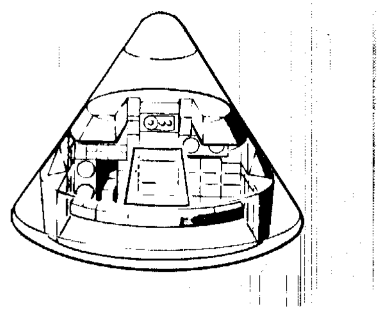
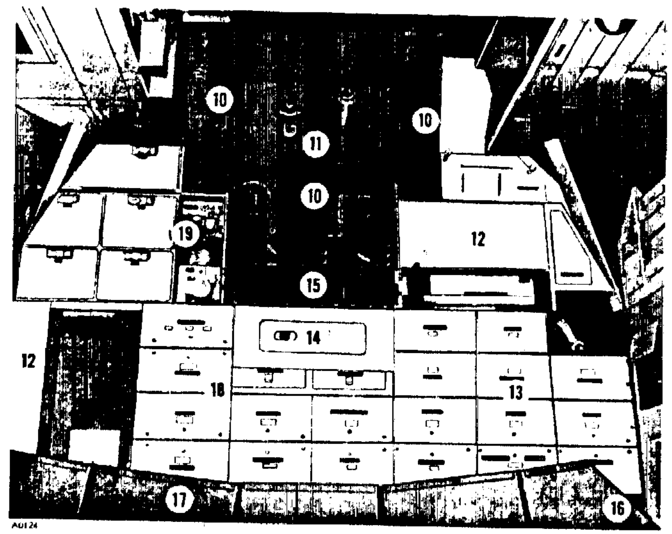
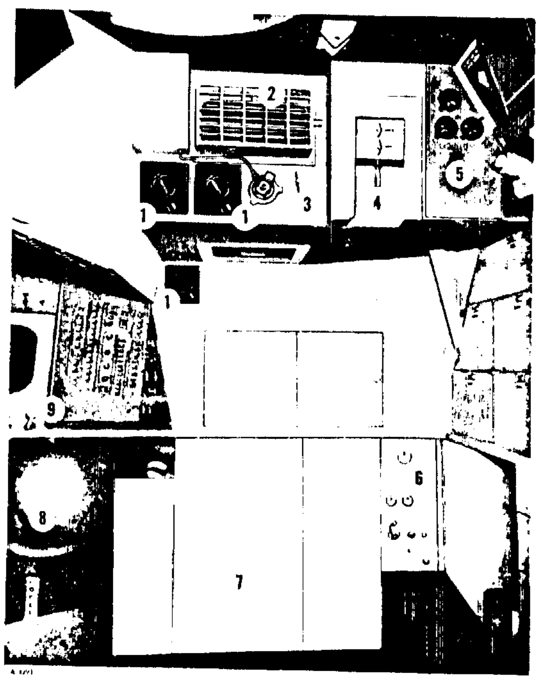
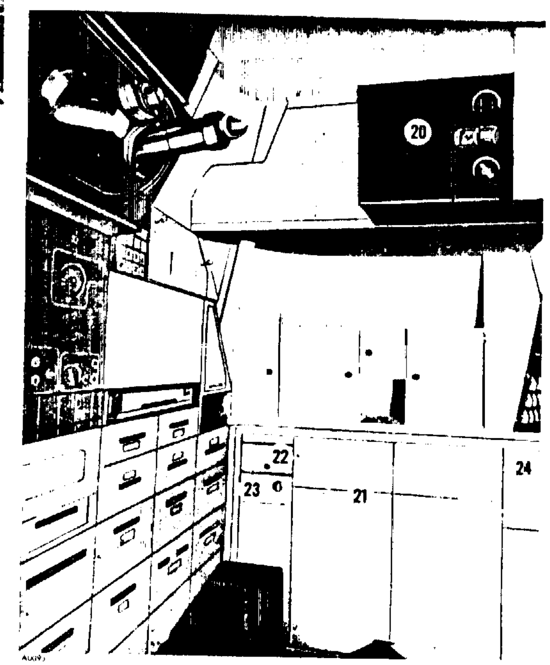


Figure 3-21. Controls and Displays - Main Display Console (Sheet 3 of 3)









EQUIPMENT IDENTIFICATION LIST

LH FORWARD EQUIPMENT BAY
1 PRESSURE SUIT CONNECTOR (ECS)
2 CABIN HEAT EXCHANGER SHUTTER (ECS)
3 CABIN TEMPERATURE CONTROL PANEL (ECS)
4 POTABLE WATER SUPPLY PANEL (ECS)
5 GMT CLOCK AND EVENT TIMERS (G&N)

LH EQUIPMENT BAY
6 OXYGEN CONTROL PANEL (ECS)
7 ENVIRONMENTAL CONTROL SYSTEM PACKAGE (BEHIND PANELS)
8 OXYGEN SURGE TANK (ECS)
9 CABIN PRESSURE RELIEF VALVE CONTROLS (ECS)

LOWER EQUIPMENT BAY
10 CONTROL PANEL (G&N)
11 G&N OPTICS
12 DATA STORAGE EQUIPMENT
13 COMMUNICATIONS MODULES
14 APOLLO GUIDANCE COMPUTER (G&N)
15 POWER SERVO ASSEMBLY (G&N)
16 BATTERY AND PYRO CONTROLS
17 CO_2 ABSORBER CARTRIDGE STOWAGE (ECS)
18 SCS MODULES
19 RATE AND ATTITUDE GYRO ASSEMBLIES (SCS)

RH FORWARD EQUIPMENT BAY
20 SYSTEMS MANAGEMENT

RH EQUIPMENT BAY
21 MISSION SEQUENCER AND SCIENTIFIC EQUIPMENT (BEHIND PANELS)
22 VACUUM CLEANER STOWAGE
23 WASTE MANAGEMENT CONTROL PANEL
24 BUS TIE AND INVERTER POWER CONTROLS

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Figure 3-22. Command Module Equipment and Storage Bays (Block I)

3-142. DOCKING SYSTEM (BLOCK II).

3-143. The purpose of the docking system (figure 3-23) is to provide a means of connecting and disconnecting the LEM/CSM during a mission and to provide a passageway to and from the LEM when it is connected to the CSM. The docking system consists of the primary structure and the necessary hardware required to support the complete docking function between the CSM and LEM.

3-144. The primary structure consists of the tunnel required for LEM ingress and egress. The system contains hatches, latches, probe, drogue, illumination, and required sealing surfaces. The docking structure will be able to withstand load imposition from the docking maneuvers and from the modal characteristics that will exist between the LEM and CSM.

3-145. Once the probe is engaged in the drogue, the probe attenuation system is activated to pull the LEM against the CSM. A crewmember in the C/M will then lock the C/M to the LEM with four semi-automatic latches, of the 12 latches around the forward circumference of the forward tunnel, providing an airtight seal. After the LEM and CSM are

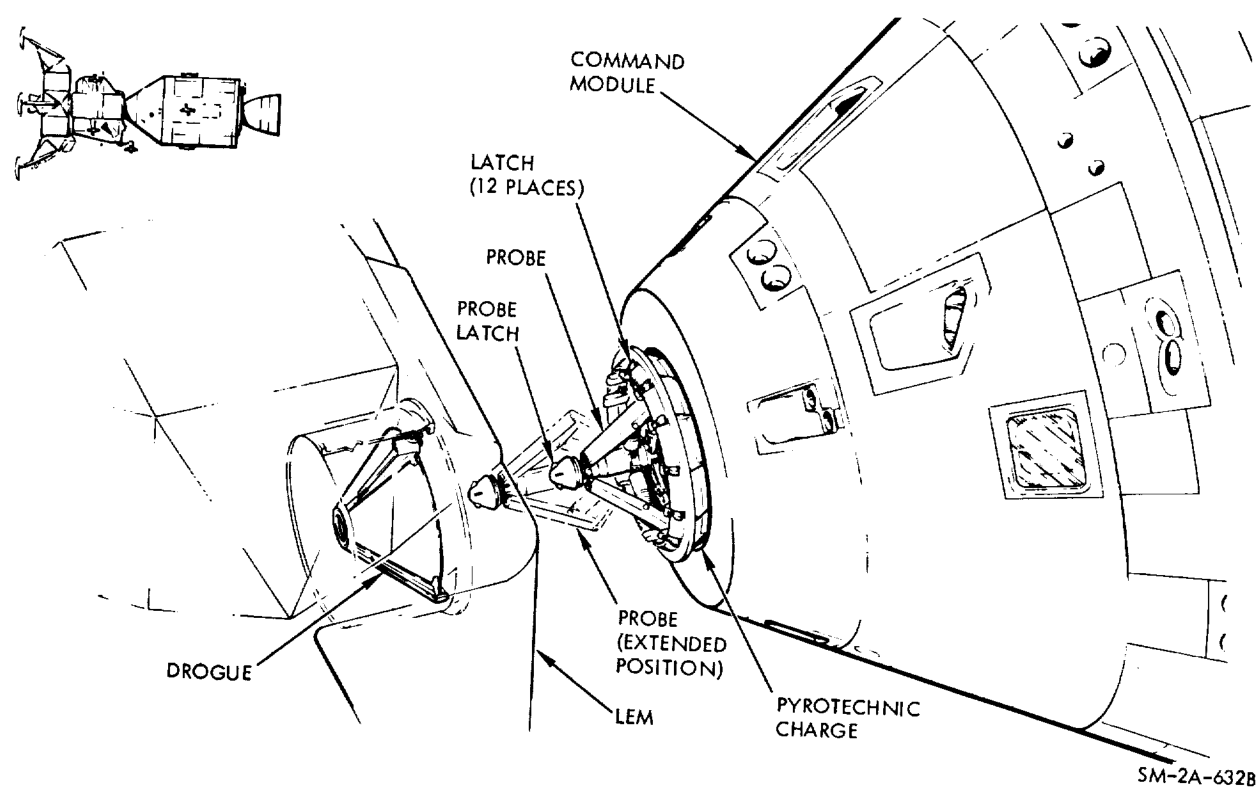


Figure 3-23. Docking System (Block II)

latched together and sealed (initial docking), the following operations are accomplished: the pressure is equalized between the LEM and C/M, the C/M forward tunnel hatch is removed, the eight remaining manual latches are locked in place (final docking), an electrical umbilical is connected to the LEM from the C/M, the probe and drogue mechanism is removed from the center of the tunnel, and the LEM hatch is removed opening a passageway between the C/M and LEM which allows transfer of equipment and crewmembers. In case a malfunction occurs which prevents docking or removal of the probe and drogue mechanism, provision has been made for emergency vehicular transfer (EVT). EVT involves transferring from one vehicle to another by way of the side hatches.

3-146. To release the LEM from the CSM for lunar landing operations, the hatches, probe, and drogue mechanisms are installed in their respective places, the 12 manual latches are released, and the probe attenuation system is electrically released allowing the LEM to separate from the CSM.

3-147. To release the LEM from the CSM in preparation for the return trip to earth, all equipment with no further use to the astronauts is placed in the LEM, including the probe and the drogue. After transfer of equipment and crewmembers, the C/M forward tunnel hatch is installed in place, resealing the C/M crew compartment. The LEM is then released from the C/M by firing a shaped pyrotechnic charge which is located around the circumference of the docking assembly.

3-148. CREWMAN ALIGNMENT SIGHT (BLOCK II).

3-149. The crewman alignment sight (CAS) is a sighting aid required for docking maneuvers after transposition of the CSM to assist the astronauts in accurately aligning the CSM with the LEM. The LEM also has an alignment sight that will be used in a similar manner after rendezvous is accomplished in lunar orbit. The CAS is a collimator-type optical device and provides the astronaut with a fixed line-of-sight attitude reference image. When viewed through the rendezvous window, the reference image appears to be the same distance away from the C/M as the LEM target.

3-150. The CAS can be mounted on the inboard side of either the LH or RH rendezvous windows, and is designed to project an optical collimated alignment image. A light source with adjustable brightness control allows for proper brightness against all exterior background lighting conditions.

3-151. When the LEM is the active docking vehicle, the CAS can be used as a backup to check the alignment of the LEM with the CSM. The CAS can also be used as a replacement for the scanning telescope when necessary.

LUNAR EXCURSION MODULE

4-1. GENERAL.

4-2. This section contains basic data concerning the lunar excursion module (LEM). Information is given to indicate configuration, function, and interface of the various components in gross terms. For more detailed LEM information, refer to the Lunar Excursion Module Familiarization Manual, LMA790-1.

4-3. The LEM, illustrated in figure 4-1, will carry two Apollo crewmembers from the orbiting CSM to the surface of the moon. The descent from lunar orbit will be powered by a gimbal-mounted rocket engine in the LEM descent stage. As the LEM nears the lunar surface, the descent engine will provide braking and hovering to allow lateral movement to a suitable landing area. The LEM will provide a base of operations for lunar explorations. Food, water, electrical power, environmental control, and communications relay will sustain the crewmembers for a period up to 48 hours. Both astronauts will explore the lunar surface at the same time, leaving the LEM unattended. Upon completion of exploration, the astronauts will return to the LEM; the LEM ascent stage will then ascend to intercept the orbiting CSM with power provided by the ascent rocket engine. The entire LEM descent stage and other nonreturnable equipment is left on the moon. Upon rendezvousing and docking with the CSM, the crew will transfer from the LEM to the CSM and, the LEM will then be jettisoned and left as a lunar satellite.

4-4. LEM CONFIGURATION.

4-5. STRUCTURE.

4-6. LEM structural components are divided into ascent-stage and descent-stage structures. (See figure 4-2.) The ascent-stage structure consists of the following components:

- Crew compartment pressure shell
- Ascent tanks and engine support
- Equipment bay
- Equipment compartment
- Electronic replaceable assembly
- Oxygen, water, and helium tanks
- Reaction control system tanks and engine supports
- Windows, tunnels, drogue mechanism, and hatches
- Docking target recess
- Interstage fittings

The descent-stage structure consists of the following components:

- Descent engine tanks and engine support
- Landing gear assembly
- Secondary oxygen, water, and helium tanks

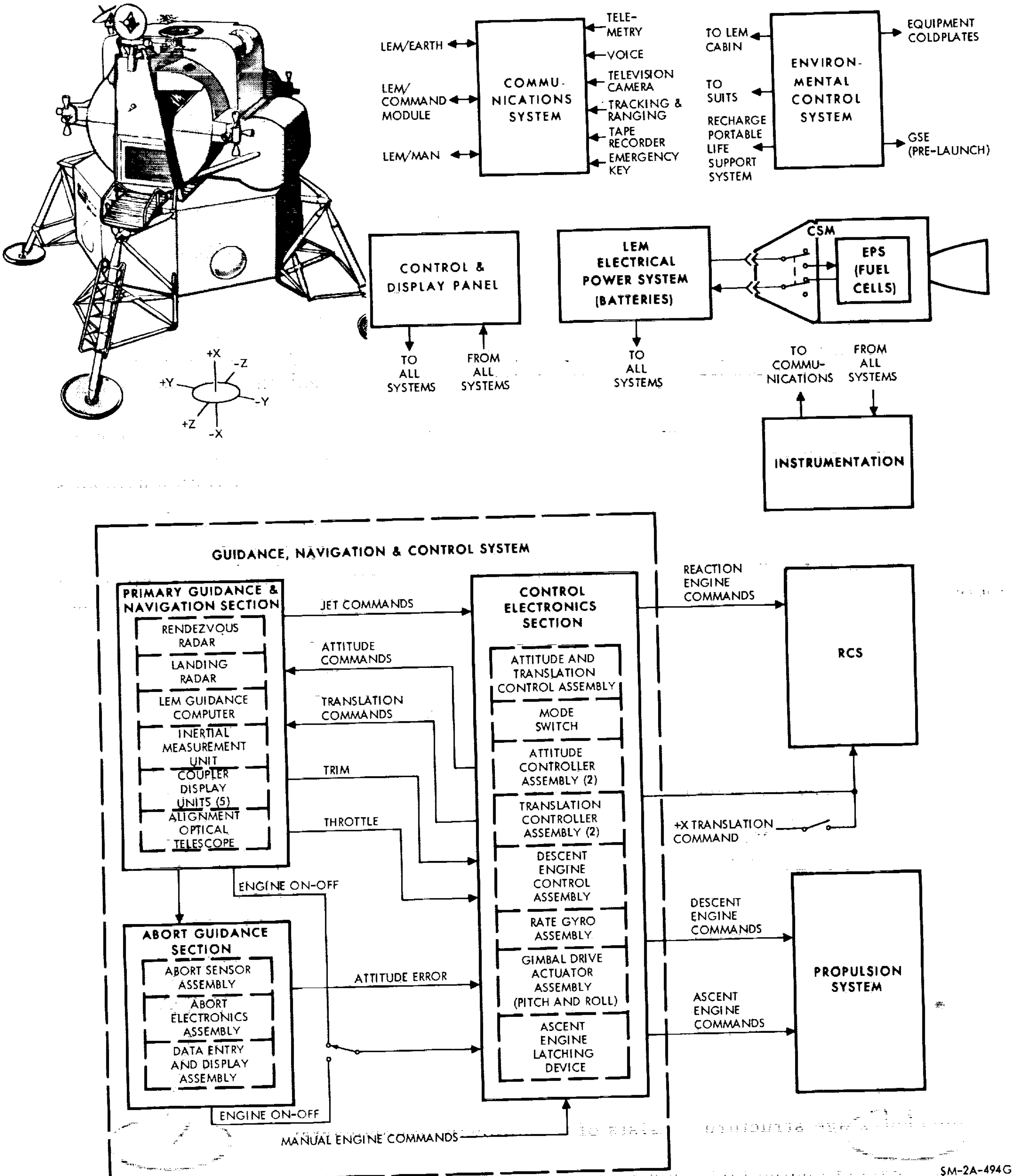


Figure 4-1. Lunar Excursion Module and Systems Block Diagram

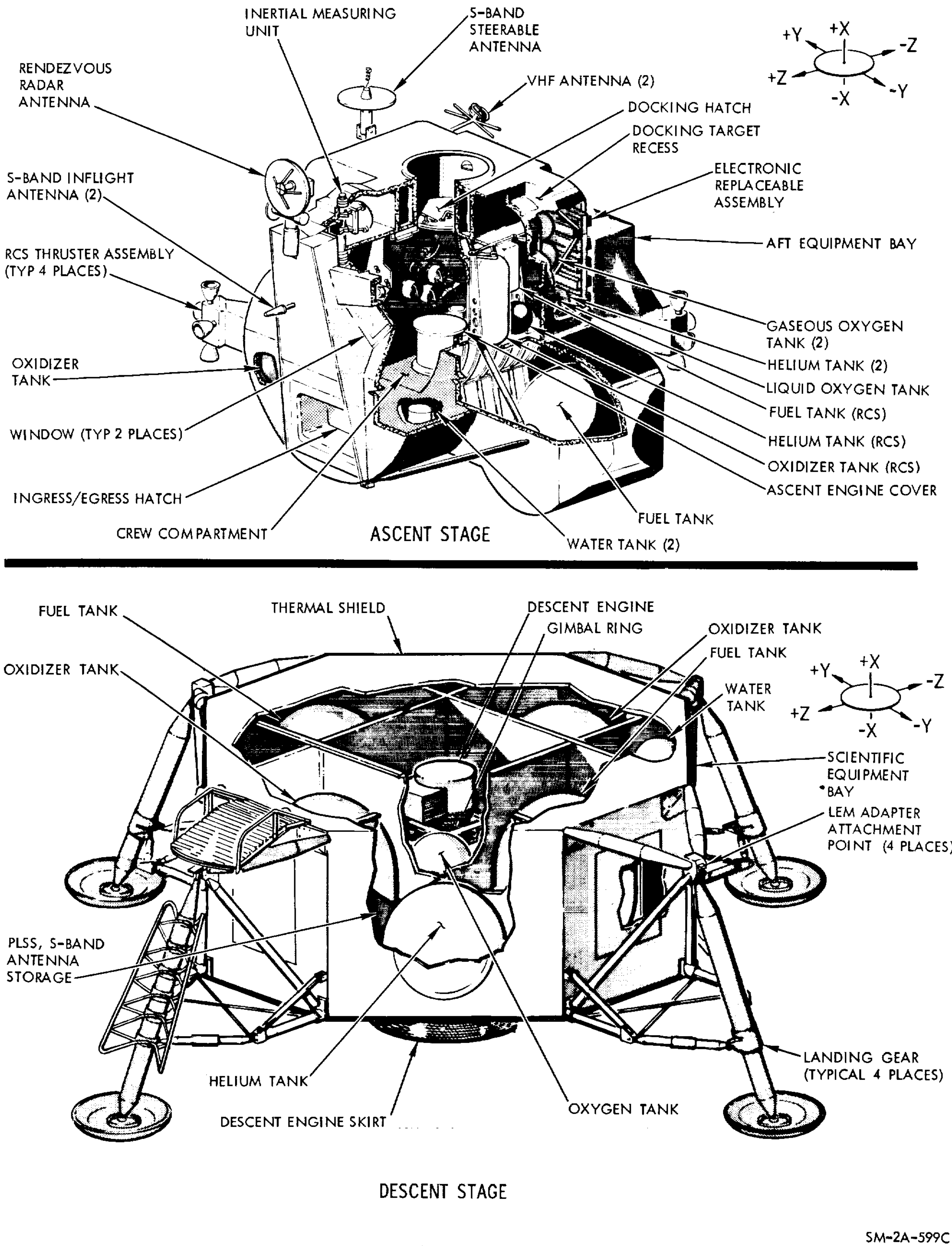


Figure 4-2. LEM Ascent and Descent Stages

- Scientific equipment bay
- Interstage fittings
- Antenna storage bay
- Battery storage bay

4-7. LEM OPERATION.

4-8. Operation of the LEM is under crew control. Controls and displays provide monitoring and allow control of the various systems. Warning and caution lights are provided on two centrally located panels. A malfunction in any of the systems will light a specific indicator, denoting the malfunctioning system. General operation of the various systems (figure 4-1) is explained in paragraphs 4-9 through 4-29.

4-9. GUIDANCE, NAVIGATION, AND CONTROL SYSTEM.

4-10. The guidance, navigation and control system is an inertial system aided by optical sighting equipment and radar. The function of the guidance, navigation, and control system is to provide and/or maintain the following:

- Position data
- Velocity
- LEM attitude
- Altitude
- Rate of ascent/descent
- Range (from command/service module)
- Range rate
- Control command

The system consists of three sections; a primary guidance and navigation section, a control electronics section, and an abort guidance section. A LEM guidance computer (LGC), inertial measurement unit (IMU), coupling display units (CDU), optical telescope, rendezvous radar, and landing radar comprise the primary guidance and navigation section.

4-11. The guidance, navigation, and control system monitors and controls the attitude of the LEM, provides guidance backup, enables automatic or manual control modes, and controls propulsion engine gimbals. Attitude error signals are generated in the primary guidance and navigation section and the abort guidance section, and routed to the control electronics section. The control electronics section provides attitude error correction signals and propulsion commands. An attitude and translation control assembly provides signal routing and mode control for the guidance, navigation, and control system. Mode control enables manual or automatic control of the LEM attitudes through the reaction control system and LEM velocity through the propulsion system. Rate gyros and an in-flight monitor provide control and display information. If the primary guidance and navigation section fails, the abort guidance section mode may be selected to take over the guidance functions. Propulsion engine, gimbal controls, and firing commands are controlled by the primary guidance and navigation section and control electronics section to ensure thrusting through the LEM center of gravity. Backup control is provided by the abort guidance section.

4-12. RADAR SYSTEMS. There are two separate radar systems which aid the guidance, navigation, and control system. The rendezvous radar provides range and range-rate information, as well as azimuth (target bearing) information, utilizing a gimbal-mounted antenna. The landing radar provides altitude and altitude change-rate information utilizing a two-position antenna. The control and display panels provide crew control of the radar system and display of the information received. A storage buffer receives the acquired information from signal conditioners; then a high-speed counter, timer by the LGC, converts the information into representative digital form which is fed into the LGC.

4-13. PROPULSION SYSTEM.

4-14. Two rocket engines provide the power required for descent and ascent. The engines use pressure-fed liquid propellants. The propellants consist of a 50:50 mixture of UDMH and hydrazine as fuel, and nitrogen tetroxide as the oxidizer. Ignition is by hypergolic reaction when the fuel and oxidizer are combined. The descent engine, fuel tanks, oxidizer tanks, and associated components are located in the LEM descent stage. Provision is made to throttle the descent engine to enable velocity control. Gimbal mounting of the engine provides hovering stability. The ascent engine is centrally mounted in the LEM, and is of fixed-thrust, nonthrottling configuration, mounted in a fixed position. The propellant supply of the ascent engine is interconnected with the reaction control system propellant supply. Control of the engines may be either manual or automatic, with automatic control maintained by the LGC through the guidance, navigation, and control system.

4-15. REACTION CONTROL SYSTEM.

4-16. LEM attitude control is provided by 16 small rocket engines mounted in four clusters. Each cluster consists of four engines mounted 90 degrees apart. The engines are supplied by two pressure-fed propellant systems. The propellants are the same as those used by the propulsion engines. The propellant supply to the reaction control system engines is also interconnected to the ascent engine propellant supply, allowing extended use of the reaction control engines. Reaction engine commands may be manual or automatic, and are applied through the guidance, navigation, and control system.

4-17. ENVIRONMENTAL CONTROL SYSTEM.

4-18. Environmental control is maintained inside the LEM cabin. Portable life support systems, in the form of backpacks, supply a controlled environment in the pressure suits to allow exploration of the lunar surface. Oxygen, water, and water-glycol are used for environmental control. Pure oxygen is stored in a tank located in the ascent stage. The pure oxygen is conditioned for use by mixing it with filtered oxygen. The descent stage contains a tank which stores additional oxygen in the super-critical (liquid or extremely cold) state. Potable water for drinking, food preparation, and the backpacks, is stored in a water tank. Temperature control of the cabin and electronic equipment is provided by a water-glycol cooling system. The coolant is pumped through the electronic equipment coldplates and heat exchangers, and filtered. Cabin temperature control is monitored by temperature sensors and maintained by a temperature controller. The portable life support system (PLSS) provide necessary oxygen, water, electrical power, and a communications link to enable the LEM crewmembers exploring the lunar surface to remain in contact with each other, the CSM and MSFN. The backpacks can be used approximately 4 hours, after which the oxygen tank must be refilled and the batteries recharged from the environmental control system.

4-19. ELECTRICAL POWER SYSTEM.

4-20. Electrical power is provided by six silver oxide-zinc, 28-vdc batteries, four in the descent stage and two in the ascent stage. Two additional batteries are provided specifically for explosive devices. The batteries will supply sufficient power to maintain essential functions of the LEM. Power distribution is provided by three buses; the commander bus, the system engineer bus, and the a-c bus. The commander and system engineer buses (28 vdc) supply power to components which must operate under all conditions. Power to all other components is provided by the a-c bus. The a-c bus is provided with 115-vac 400-cps power by one of two inverters selected by a crewmember. The two electroexplosive device batteries provide power to fire explosive devices for the landing gear uplock, stage separation, and helium pressurizing valves in the propulsion and reaction control systems.

4-21. COMMUNICATIONS.

4-22. Communications aboard the LEM are divided into three systems, listed as follows:

- LEM-earth system
- LEM-command module system
- LEM-crewmember system

The LEM-earth system will provide telemetry, television, voice, taped playback, hand-key, and transponder communication to earth. Return from earth will be in the form of voice and digital up-data. The LEM-C/M system will provide voice communications between the orbiting C/M and LEM. PCM telemetry data at 1.6 kilabits per second can be transmitted from the LEM to the command module. The LEM-crewmember system provides intercommunication for the LEM crew, and voice suit telemetry communication is provided by the backpacks when one crewmember is on the lunar surface conducting explorations.

4-23. INSTRUMENTATION.

4-24. Operational instrumentation senses physical data, monitors the LEM subsystems during the unmanned and manned phases of the mission, prepares LEM status data for transmission to earth, stores time-correlated voice data as required, and provides timing frequencies for the other LEM subsystems. The instrumentation subsystem consists of sensors, signal conditioning electronics assembly, caution and warning electronics assembly, pulse code modulation and timing electronics assembly, and the data storage electronics assembly.

4-25. CONTROL AND DISPLAY PANELS.

4-26. The controls and display panels contain controls, monitoring instruments, and warning indicators to enable the crewmembers to maintain full knowledge of the status of various systems. Manual overrides allow the crewmembers to compensate for any deviations not allowed in automatic systems operation, or to take over a malfunctioning operation.

4-27. CREW PROVISIONS.

4-28. The crew provisions consist of miscellaneous equipment necessary to support two crewmen in the descent, the 24- to 48-hour exploration, and the ascent phases. The items included are listed as follows:

- Extravehicular mobility unit (Includes space suits, garments, and PLSS)
- Astronaut supports and restraints
- Lighting
- First-aid kit
- Food storage and water dispensing
- Waste management section
- Medical kit

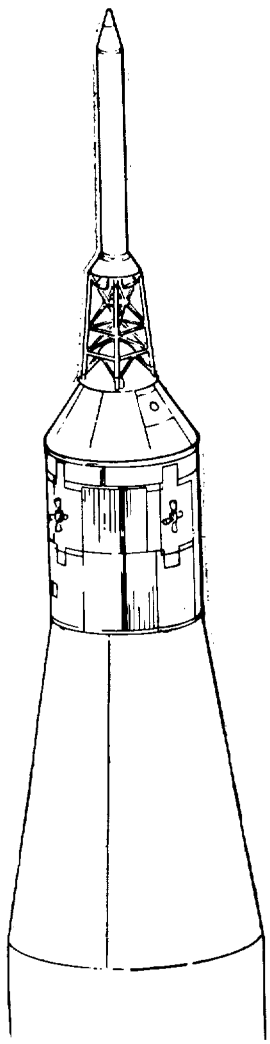
4-29. SCIENTIFIC INSTRUMENTATION.

4-30. Scientific instrumentation will be carried to the lunar surface aboard the LEM to enable the crewmembers to acquire samples and data concerning the lunar environment. A list of typical instrumentation to be used is as follows:

- Lunar atmosphere analyzer
- Gravitometer
- Magnetometer
- Penetrometer
- Radiation spectrometer
- Specimen return container
- Rock and soil analysis equipment
- Seismograph
- Soil temperature sensor
- Self-contained telemetering system
- Camera
- Telescope

As additional data concerning the lunar environment become available, this list will be altered.

APOLLO SPACECRAFT MANUFACTURING



5-1. GENERAL.

5-2. This section describes the manufacturing fabrication, assembly, subsystems installation, and functional checkout of Apollo spacecraft structures and systems. Manufacturing techniques and processes utilized, incorporate features applicable to the special requirements of the Apollo spacecraft. Final assembly, subsystems installation, and functional checkout is performed in environmentally controlled cleaning rooms, providing control of humidity, temperature, and sources of contamination.

5-3. SPACECRAFT MAJOR ASSEMBLIES.

5-4. The spacecraft is comprised of four major assemblies consisting of the launch escape system, the command module, service module, and spacecraft LEM adapter (SLA). The SLA houses the lunar excursion module.

5-5. LAUNCH ESCAPE SYSTEM STRUCTURE.

5-6. The launch escape system structure (figure 5-1) consists of a nose cone, canard assembly, pitch control motor, tower jettison motor, launch escape motor, skirt structure assembly, tower structure assembly, and hard and soft boost protective covers. The entire system is 33 feet long.

5-7. The canard assembly is made from Inconel nickel and stainless steel skins riveted together. The tower structure assembly is a fusion-welded, titanium tubing structure with fittings at each end for attachment to the skirt structure assembly and the

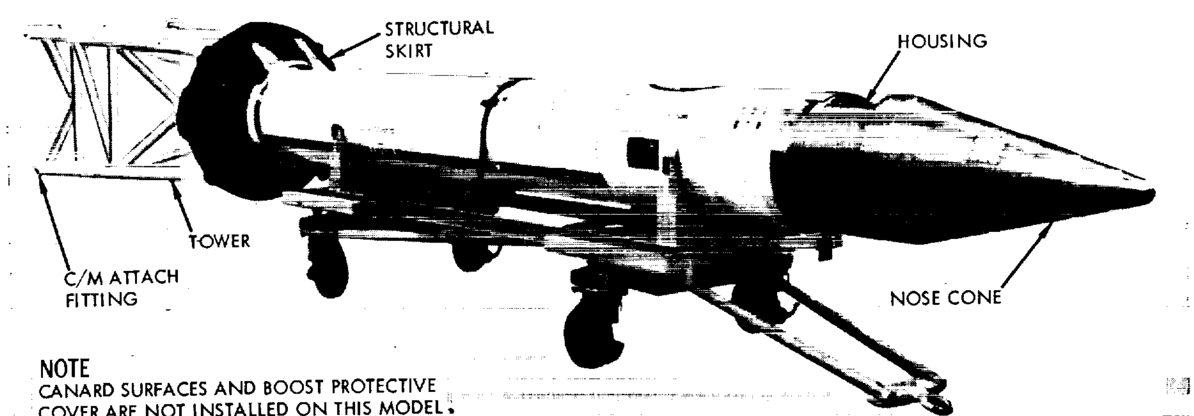


Figure 5-1. Launch Escape System Structure

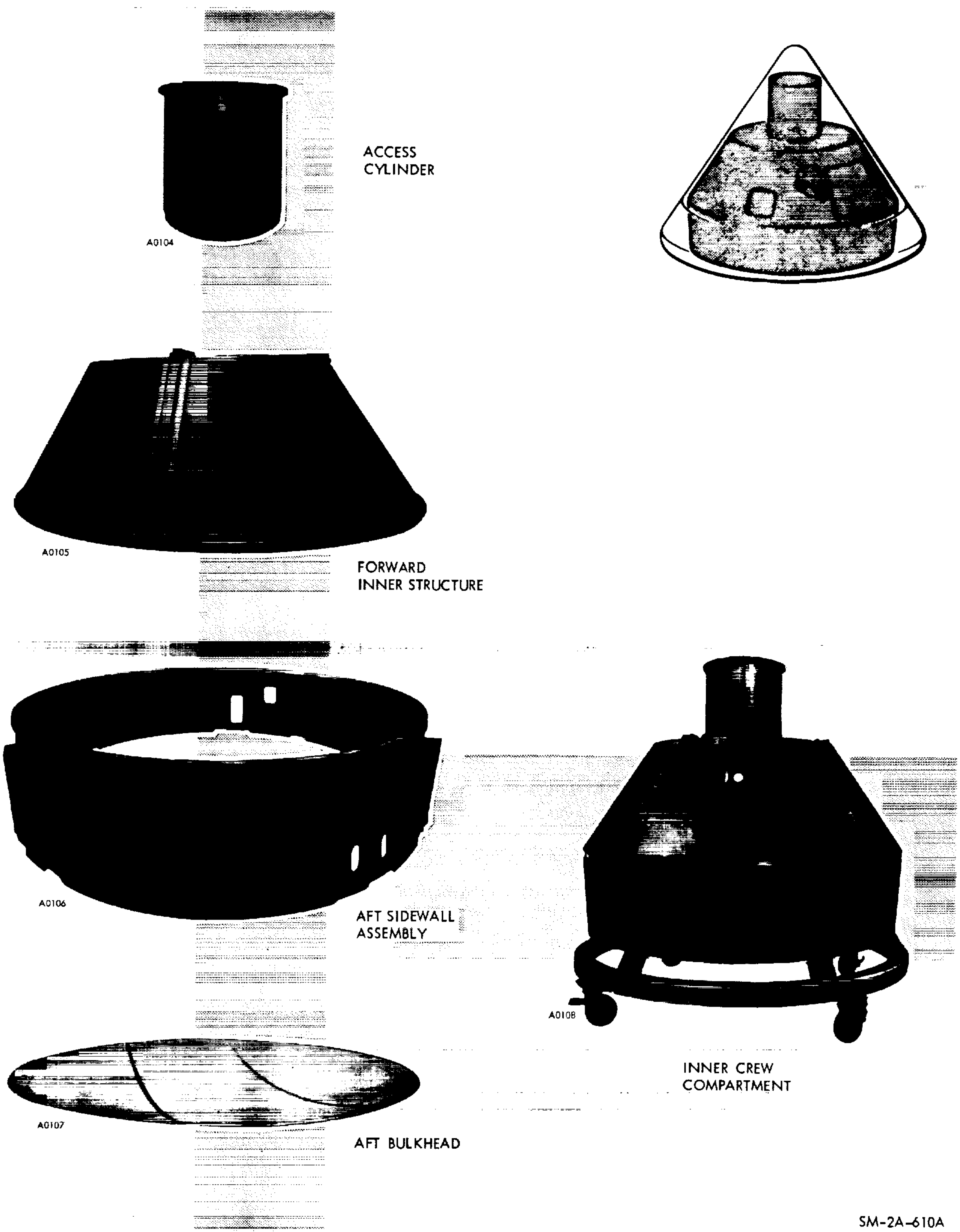


Figure 5-2. Command Module Inner Crew Compartment Structure

command module release mechanism. The ballast enclosure and pitch control motor structure assemblies are fabricated from nickel alloy steel sheet metal skins, which are riveted to ring bulkheads and frames. The skirt assembly is made from titanium and is welded and riveted during construction. The boost protective covers, constructed of glass cloth, phenolic honeycomb, and ablative cork, are fastened to the bottom of the tower.

5-8. COMMAND MODULE STRUCTURE.

5-9. The basic structure of the command module consists of a nonpressurized outer heat shield and a pressurized inner crew compartment. Bolted frame assemblies secure the heat shield to the aft compartment while I-beam stringers are used to mechanically fasten the inner and outer structures in the forward crew compartment area. A two-layer microquartz fiber insulation is installed between the inner and outer structures. Ablative materials are applied to the outer heat shield to protect the C/M against aerodynamic heating during entry into the atmosphere of the earth.

5-10. INNER CREW COMPARTMENT STRUCTURE. The inner crew compartment consists basically of a forward section containing an access cylinder welded to the forward bulkhead and cone, and an aft section containing the sidewall and bulkhead. (See figure 5-2.) These two sections are welded into subassemblies, then honeycomb bonded, trimmed, butt-fusion welded (figure 5-3), and the closeout area is filled with aluminum honeycomb material. Face sheets of aluminum alloy overlap the two sections and are bonded in place. Secondary structure equipment bays, housing the various subsystems and storage areas, are located within the inner crew compartment. Refer to section III for a description of the systems installed in the C/M.

5-11. HEAT SHIELD STRUCTURE. The heat shield structure (figure 5-4) of the command module consists of the apex cover, forward heat shield, crew compartment heat shield, and aft heat shield. The apex cover, used on Block I S/C, is replaced with the C/M portion of the docking system on Block II S/C.

5-12. The forward heat shield consists of four honeycomb panels, two rings, and four launch escape tower leg wells. The panels are placed in a jig singly and trimmed longitudinally. The tower leg wells are installed, trimmed, and welded. The panels are then installed in a jig which accommodates all four panels, trimmed longitudinally, and butt-fusion welded. The welded panels and the rings are placed in another jig for circumferential trim, and the rings are then welded to the top and bottom of the panels. The completed assembly is fit-checked to the crew compartment and aft heat shields, and then removed for the application of ablative material.

5-13. The crew compartment heat shield is formed from steel honeycomb panels and rings. The panels are joined together by machined edge-members which provide door-opening lands and are attached to the inner crew compartment structure. The panels and rings are installed in a series of jigs for assembly, trimming, and welding. The welded sections are then placed in a large fixture for precision machining of the top and bottom rings. The assembly is fit-checked with the inner crew compartment, forward heat shield, and aft heat shield, and then removed for the application of ablative material.

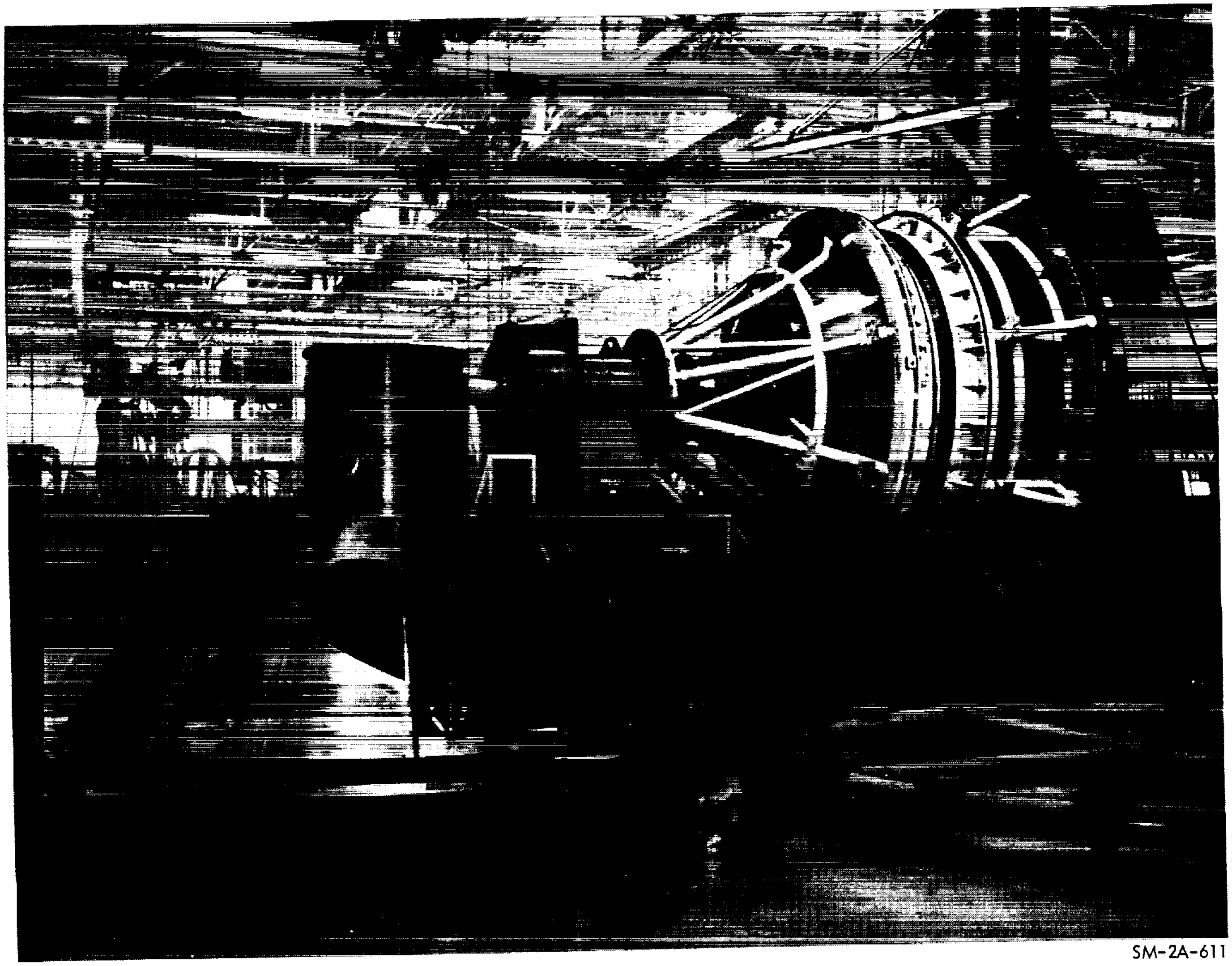


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Figure 5-3. Trim and Weld Closeout Operation

5-14. The aft heat shield assembly consists of four brazed honeycomb panels joined laterally by fusion welds and attached to a 360-degree machined ring by spot-welded sheet metal fairings, using conventional mechanical fasteners. Holes for the inner and outer C/M component attach points and the tension tie locations are cut through the assembly by (tre-pan) machining. The completed aft heat shield is fit-checked with the crew compartment heat shield prior to the application of ablative material.

5-15. SERVICE MODULE STRUCTURE.

5-16. The service module consists basically of a forward and aft bulkhead, radial beams and outer panels, and is constructed primarily of aluminum alloy which is honeycomb bonded. The eight outer panels are aluminum honeycomb bonded between aluminum face sheets. The forward bulkheads of the Block II S/M will be constructed of sheet metal with the aft bulkhead remaining aluminum-bonded honeycomb. Six radial beams, which divide the cylinder into compartments, are machined and chem-milled to reduce weight in non-critical stress areas. Beams, bulkheads, and support shelves form the basic structure. (See figure 5-5.)

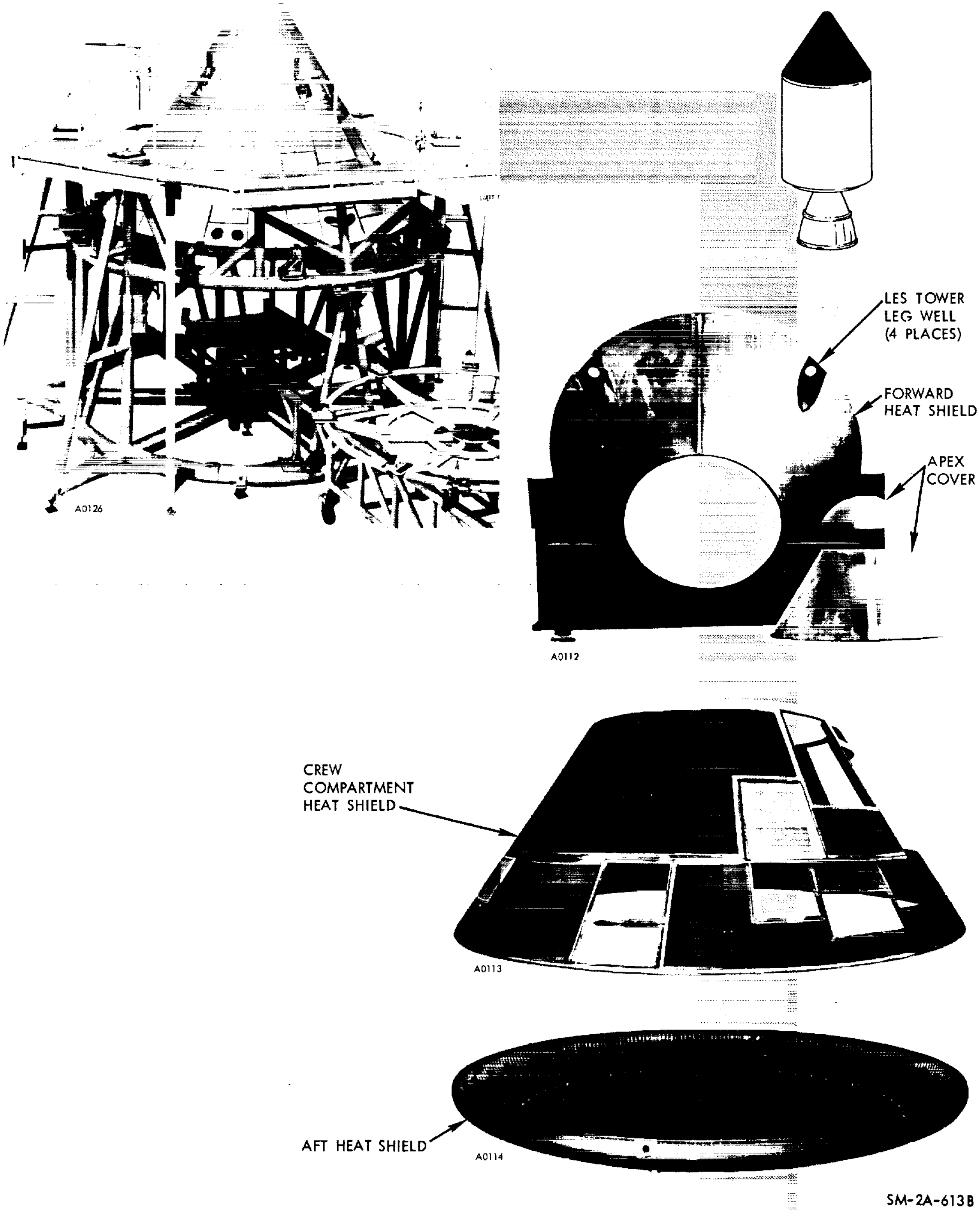


Figure 5-4. Command Module Heat Shield Structure

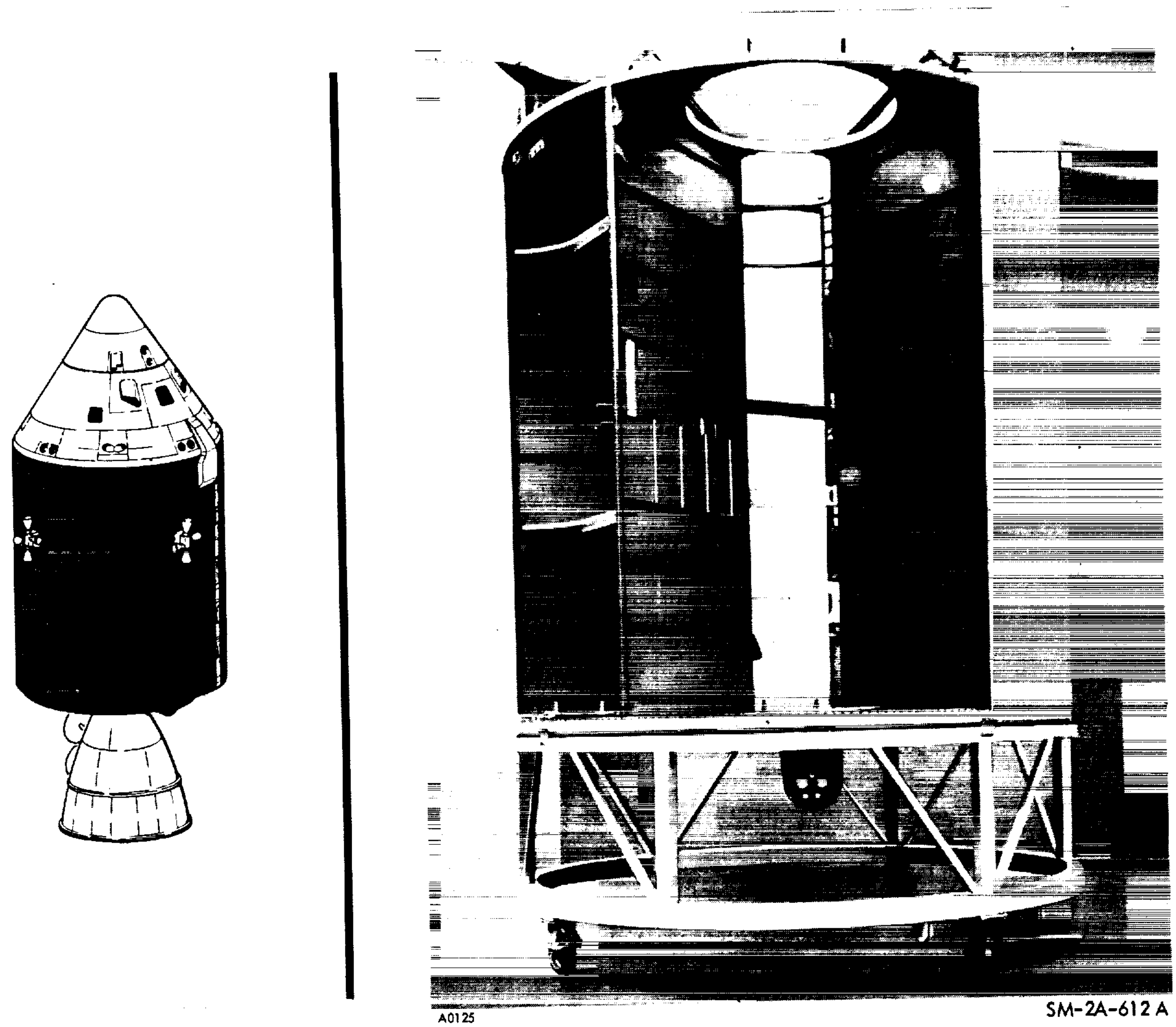


Figure 5-5. Service Module Structure

5-17. The fuel and oxidizer tanks, hydrogen and oxygen tanks, fuel cells, reaction control system, service propellant system, antenna equipment, electrical power system, and part of the environmental control system, are housed in the service module.

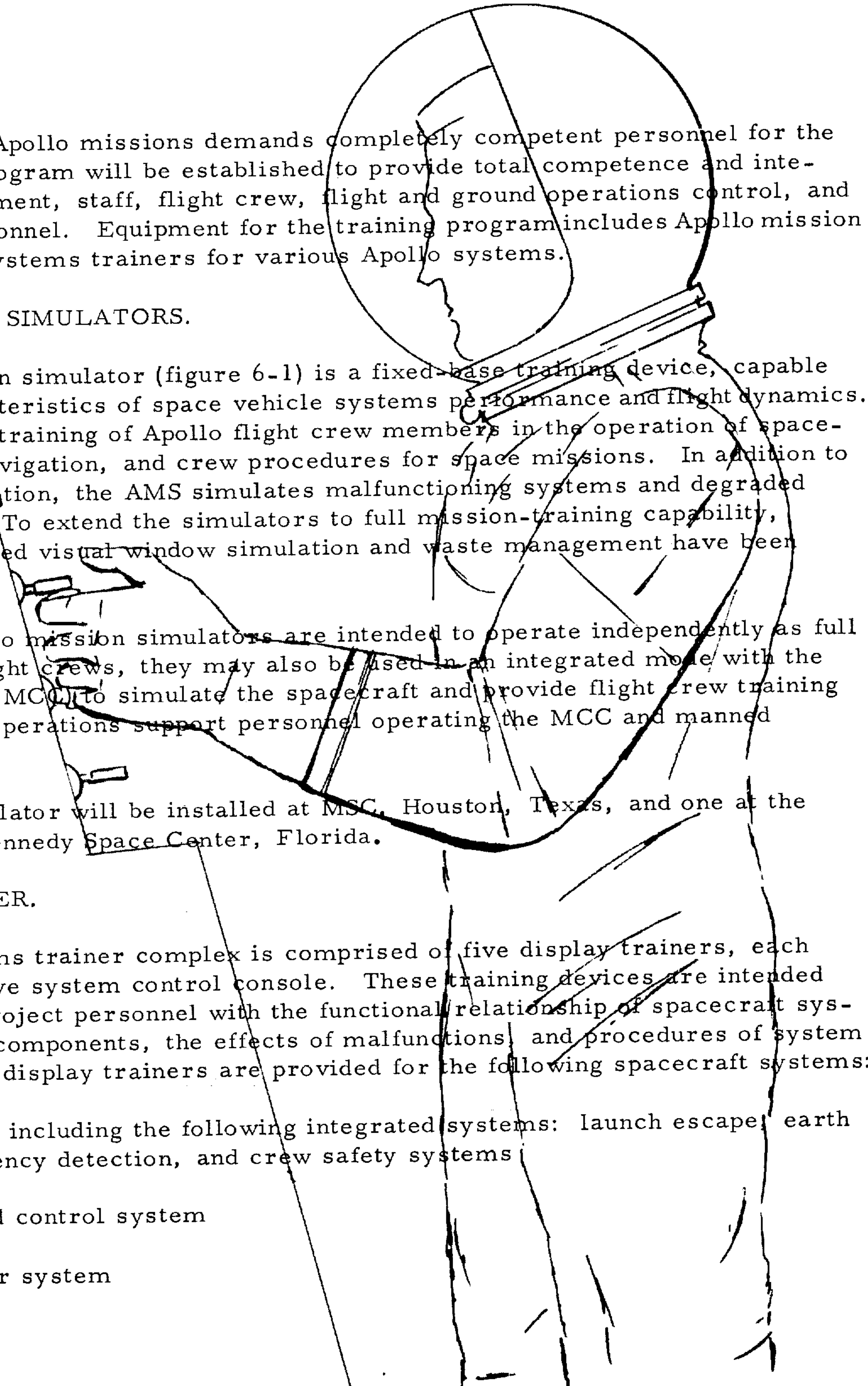
5-18. SPACECRAFT LEM ADAPTER.

5-19. The spacecraft LEM adapter (SLA) is a truncated cone, constructed of bonded aluminum honeycomb, which connects the S/M with the S-IVB instrument unit and houses the LEM. The adapter is 28 feet in length, 12.8 feet in diameter at the forward end, and 21.6 feet in diameter at the aft end. The SLA consists of eight 2-inch-thick bonded aluminum honeycomb panels, which are joined together with riveted inner and outer doublers. Linear-shaped charges will be installed on four of the panels, which are hinged at the aft end, to provide a means of exposing the LEM and separating the S/M from the SLA.

5-20. MODULE MATING AND FINAL ASSEMBLY.

5-21. Upon completion of structural assembly, the modules are cleaned and sent to a cleanroom for installation and checkout of all systems. The modules are then mated for fit-check and alignment to ensure conformance to design. Alignment is checked optically with theodolites, sight levels, or autocollimators. Each module is also given a weight and balance check to determine its center-of-gravity. Following completion of individual and combined systems checkout, a detailed prelaunch integrated systems check is performed. After assurance that all systems perform according to design criteria, the modules are demated, packaged, and shipped to the designated test site.

APOLLO TRAINING EQUIPMENT



6-1. GENERAL.

6-2. The nature of the Apollo missions demands completely competent personnel for the program. A training program will be established to provide total competence and integration between management, staff, flight crew, flight and ground operations control, and test and operations personnel. Equipment for the training program includes Apollo mission simulators and Apollo systems trainers for various Apollo systems.

6-3. APOLLO MISSION SIMULATORS.

6-4. The Apollo mission simulator (figure 6-1) is a fixed-base training device, capable of simulating the characteristics of space vehicle systems performance and flight dynamics. The simulator provides training of Apollo flight crew members in the operation of spacecraft systems, space navigation, and crew procedures for space missions. In addition to normal spacecraft operation, the AMS simulates malfunctioning systems and degraded systems performance. To extend the simulators to full mission-training capability, telemetry data link, added visual window simulation and waste management have been added.

6-5. Although the Apollo mission simulators are intended to operate independently as full mission trainers for flight crews, they may also be used in an integrated mode with the mission control center (MCC) to simulate the spacecraft and provide flight crew training in conjunction with the operations support personnel operating the MCC and manned space-flight network.

6-6. One mission simulator will be installed at MSC, Houston, Texas, and one at the Eastern Test Range, Kennedy Space Center, Florida.

6-7. SYSTEMS TRAINER.

6-8. The Apollo systems trainer complex is comprised of five display trainers, each having its own respective system control console. These training devices are intended to familiarize Apollo project personnel with the functional relationship of spacecraft systems, subsystems and components, the effects of malfunctions, and procedures of system management. The five display trainers are provided for the following spacecraft systems:

- Sequential flow, including the following integrated systems: launch escape, earth landing, emergency detection, and crew safety systems
- Stabilization and control system
- Electrical power system

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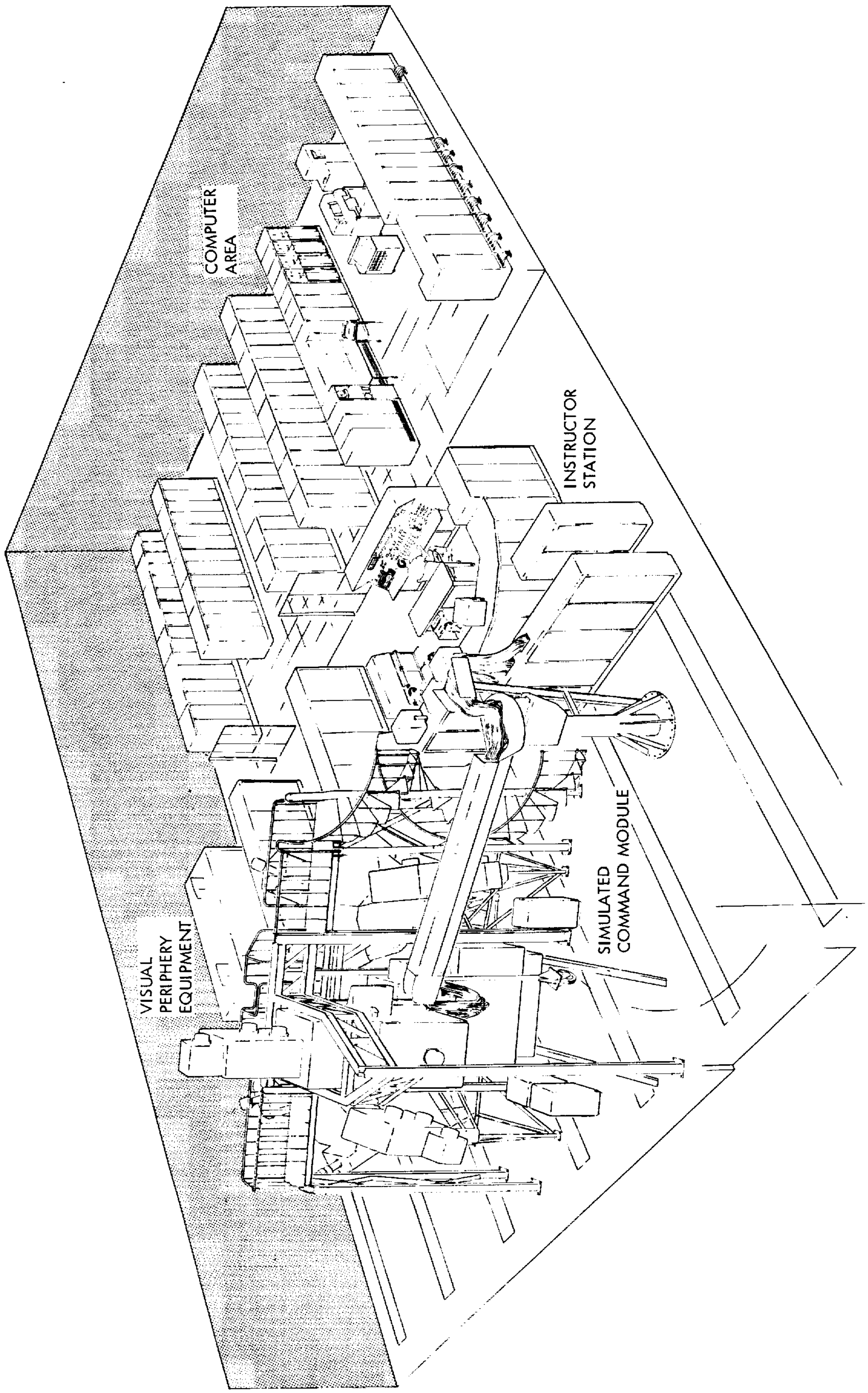


Figure 6-1. Apollo Mission Simulators Installation

- Environmental control system
- Spacecraft propulsion systems (service propulsion and reaction control systems)

6-9. The sequential flow display trainer is capable of accurately displaying a schematic diagram of all sequential operations of the launch escape, earth landing, emergency detection, and crew safety systems. Two panels are incorporated in the trainer to demonstrate normal launch, pad abort, high-altitude abort, early-mission termination, and normal earth landing sequence. Sequence malfunction simulation is limited to circuit interruption, disrupting component operation presented on the panel displays.

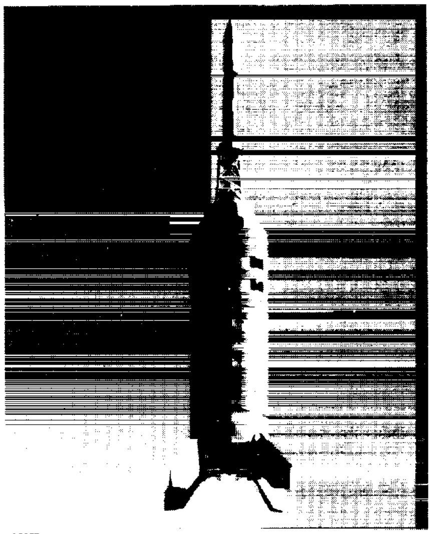
6-10. The stabilization and control display trainer presents a functional flow diagram depicting system operation including various switching functions. Simulated spacecraft panels are incorporated in the trainer to simulate the normal operation and malfunction of system components which control and stabilize the spacecraft flight attitude.

6-11. The electrical power system display trainer accurately depicts a schematic flow diagram showing d-c and a-c power distribution to the main buses, including fuel cell and cryogenic storage system flow diagrams. Panels on the trainer demonstrate fuel cell operation, inverter operation, battery recharging, bus switching, and systems management through use of spacecraft panel monitors. Malfunction inputs provide high or low voltage, overload, reverse current, and out-of-tolerance fuel cell monitoring conditions.

6-12. The environmental control system display trainer utilizes two panels to depict, by flow diagram, the pressure suit supply system, oxygen supply system, oxygen cryogenic input system, water-glycol system, and water system during normal ascent operation from the launch pad, while in space, and during entry. Simulated spacecraft panel monitors will be activated to show the pressures and temperatures for the operating modes. A malfunction capability is incorporated in the trainer to indicate high- or low-system parameters and, emergency conditions such as cabin pressure loss and contaminated potable water.

6-13. The propulsion system display trainer utilizes three display panels to present plumbing diagrams of the command module reaction control system, service module reaction control system, and service propulsion system in both manual and automatic modes of operation. Malfunction switches are incorporated in the panels to demonstrate visual malfunction of system components as indicated on spacecraft panel monitors.

APOLLO TEST PROGRAM



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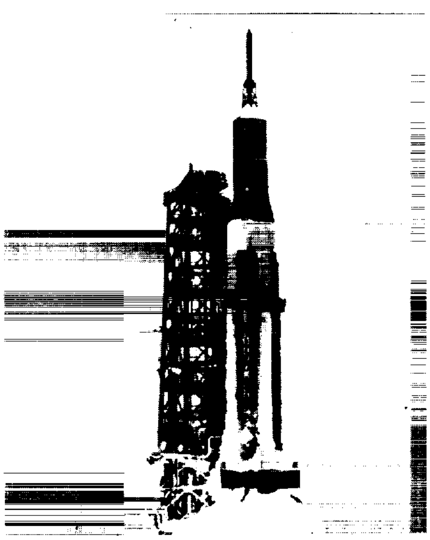

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7-1. GENERAL.

7-2. This section delineates the test program for the development of Apollo spacecraft. The development program is divided essentially into two blocks, with three interrelated phases: Block I boilerplate and spacecraft missions, Block II spacecraft missions, and propulsion system testing for both blocks. A description of ground support equipment categories and completed Apollo missions is also presented.

7-3. Boilerplates are research and development vehicles which simulate production spacecraft in size, shape, structure, mass, and center of gravity. Each boilerplate will be equipped with instrumentation to record mission parameter data for engineering review and evaluation. The data gained from the testing of boilerplate configurations will be used in determining production spacecraft flight parameters.

7-4. Spacecraft are production vehicles. These vehicles will incorporate any modification, flight profile change, and operating technique revisions deemed necessary as a result of boilerplate mission evaluation. Spacecraft configurations will vary in order to meet interface requirements of Saturn V, Saturn IB, and Little Joe II boosters. Variations will be made in the spacecraft adapters and the inserts required to satisfy booster interface. C/M and S/M size will remain constant.

7-5. Propulsion system testing will be accomplished with propulsion system test fixtures. The fixtures are not ground support equipment items, but are unique test platforms for the spacecraft propulsion system. The fixtures are fully instrumented to record engine and propellant system operation through varied operating ranges.

7-6. SPACECRAFT DEVELOPMENT.

7-7. Spacecraft development includes tests and vehicles used for the development of manned spacecraft. The relation between test vehicles, boilerplates, spacecraft, and the Apollo spacecraft development program is shown in figure 7-1.

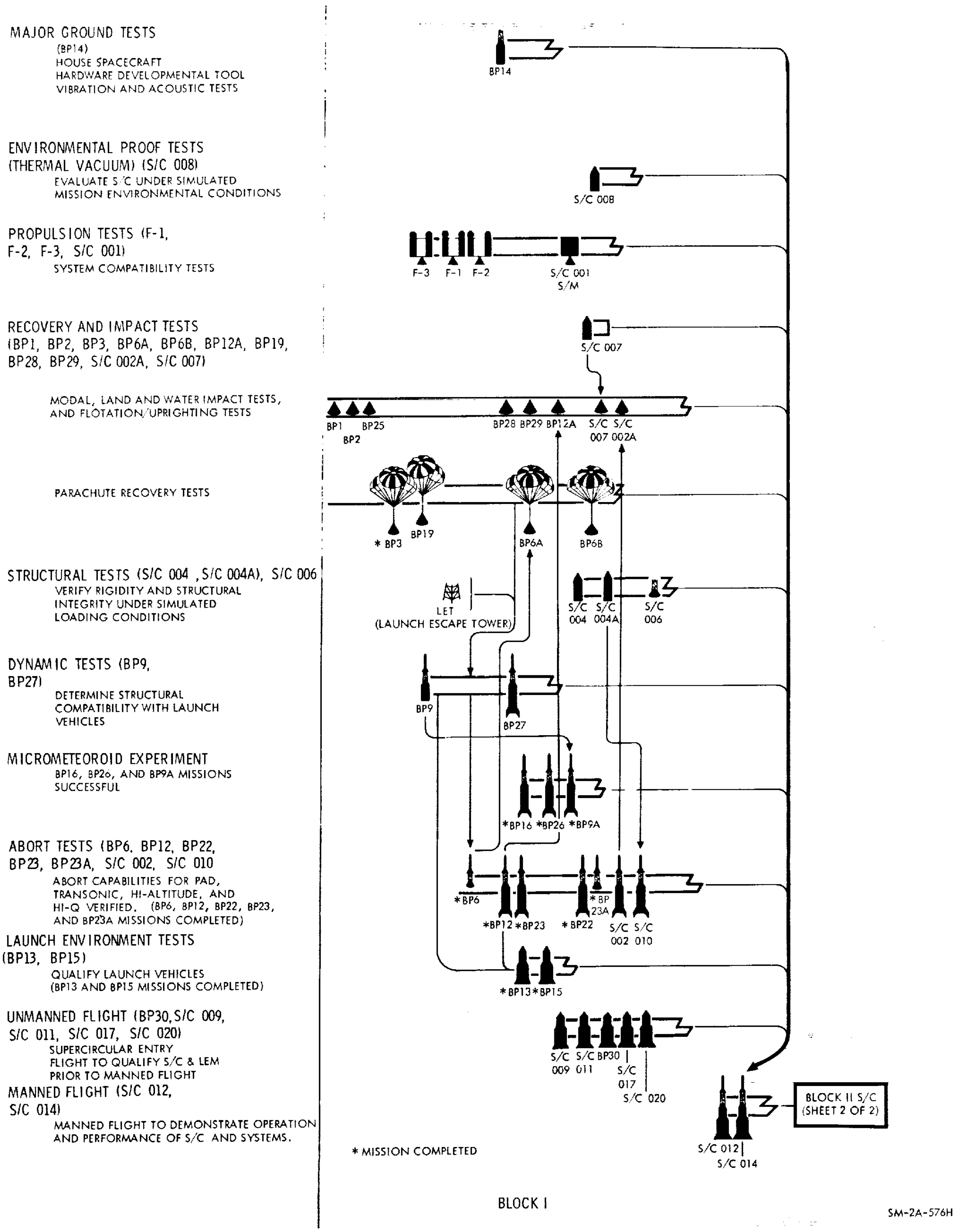

Figure 7-1. Apollo Spacecraft Development Program (Sheet 1 of 2)

MAJOR GROUND TEST VEHICLES
(S/C 2H-1)
HOUSE SPACECRAFT FOR SYSTEMS PERFORMANCE VERIFICATION

RECOVERY TEST VEHICLES
(S/C 2S-1 AND S/C 007A)
WATER AND LAND, IMPACT/FLOTATION AND POST LANDING TESTS

STATIC TESTS
(S/C 2S-2)
S/M FOR STATIC STRUCTURAL TESTS

ENVIRONMENTAL PROOF VEHICLE
(S/C 2TV-1)
THERMAL VACUUM TESTS

MANNED FLIGHT
(S/C 101 THROUGH 112)
EARTH ORBITAL AND LUNAR LANDING MISSION

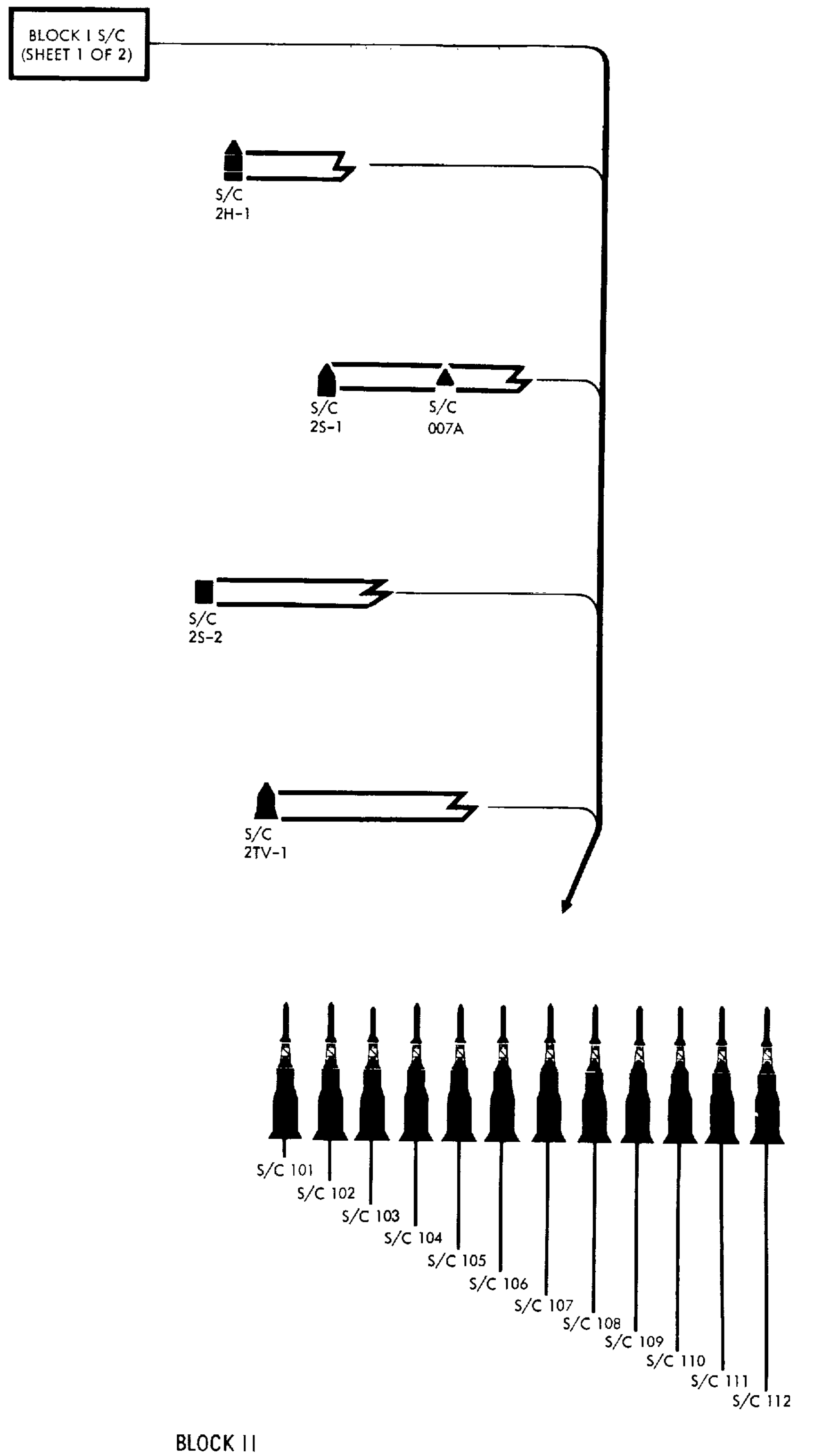

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Figure 7-1. Apollo Spacecraft Development Program (Sheet 2 of 2)

SYSTEMS CONFIGURATION LEGEND

- C - COMPLETE
- P - PARTIAL
- R - R & D INSTRUMENTATION EQUIPMENT ONLY
- S - SIMULATED OR INERT
- SP - SPECIAL
- M1 - CONTROL PROGRAMER
- M2 - APOLLO MISSION PROGRAMER FOR UNMANNED MISSIONS

NOTE: THE INFORMATION IN THIS CHART IS BASED ON THE LATEST PROGRAM PLAN. NO ATTEMPT IS MADE TO SHOW IMPENDING CHANGES. THE CHARTS WILL BE UPDATED TO REFLECT OFFICIAL PROGRAM CHANGES DURING SUBSEQUENT REVISIONS OF THIS FAMILIARIZATION MANUAL.

	BOILER-PLATES	TEST SITES	MISSIONS OR PURPOSE	STRUCTURE (Tower, Command Module, Service Module, Boost Protective Cover)	ADAPTER (S-IV, S-IVB, Extension, Special)	LAUNCH VEHICLES (Saturn I, Saturn IB, Saturn V, Little Joe II)	LAUNCH ESCAPE SYSTEM (LES) (Launch Escape Motor, Pitch Control Motor, Jettison Motor, Abort Sequencer, Canard)	SERVICE PROPULSION SYSTEM (SPS)	SEPARATION SYSTEM (LES - CM, CM - SM, SM - Adapter, Adapter Booster)	EARTH LANDING SYSTEM (ELS) (Parachute, Recovery Aids, Recovery Sequencer, Flotation Equip.)	GUIDANCE & NAVIGATION SYSTEM (G&N)	STABILIZATION & CONTROL SYSTEM (SCS)	REACTION CONTROL SYSTEM (RCS) (Command Module, Service Module)	ENVIRONMENTAL CONTROL SYSTEM (ECS) (CM, SM)	ELECTRICAL POWER SYSTEM (EPS) (CM, SM)	COMMUNICATIONS SYSTEM	DEEP SPACE (S-BAND) EQUIP.	INSTRUMENTATION SYSTEM	CONTROLS AND DISPLAYS	APOLLO MISSION PROGRAMER	CREW COUCHES	SUPPORT & RESTRAINT	CREW SYSTEM	ANTHROPOMORPHIC DUMMIES	LUNAR EXCURSION MODULE
MAJOR GROUND TEST VEHICLES	BP-14	DOWNEY, CALIF.	HOUSE SPACECRAFT NO. 1 (HARDWARE DEVELOPMENT TOOL)	C C	C		C	S	P P S S	C C C	C	C	P P	C P	P P	C	P	C	C		C	P	P		
EARTH RECOVERY AND IMPACT TEST VEHICLES	BP-1	DOWNEY, CALIF.	COMMAND MODULE FOR LAND AND WATER IMPACT TESTS	C														P			S	S		S	
	BP-2	DOWNEY, CALIF.	LAND AND WATER IMPACT TESTS, AND UPRIGHTING SYSTEM DEVELOPMENT TESTS	C						C								P							
	BP-6A (REFURBISHED BP-6)	EL CENTRO, CALIF.	PARACHUTE RECOVERY TESTS	C						C C P					R	R		P				P			
	BP-6B (REFURBISHED BP-6A)	EL CENTRO, CALIF	PARACHUTE RECOVERY TESTS	C						C C P					R	R		R							
	BP-3	EL CENTRO, CALIF.	COMMAND MODULE FOR PARACHUTE RECOVERY TESTS (DESTROYED)	C						C C					R	R		P				P			
	BP-12A (REFURBISHED BP-12)	DOWNEY, CALIF.	WATER IMPACT TEST	C						P															
	BP-19	EL CENTRO, CALIF.	SAME AS BP-3	C						C C P					R	R		P							
	BP-25	MSC	WATER RECOVERY AND HANDLING DEVELOPMENT	C																					
	BP-28	DOWNEY, CALIF.	LANDING IMPACT TEST	C						S	S	S	S	S	S	S	S	S	S & R		S & C	S & C	S	S & C	
	BP-29	MSC	FLOTATION TESTS AND QUALIFICATION OF BLOCK I UPRIGHTING SYSTEM	C						P C					P	P									
ABORT TEST VEHICLES	BP-6 (MISSION COMPLETE)	WSMR	PAD ABORT VEHICLE	C C	C		C C C C		C	C C					R & P	R		P							
	BP-12 (MISSION COMPLETE)	WSMR	VEHICLE FOR TRANSONIC ABORT USING LITTLE JOE II	C C C	C	C	C C C C		C C	C C				SP	R & P	R		P							
	BP-22 (MISSION COMPLETE)	WSMR	VEHICLE FOR HI-ALTITUDE ABORT VERIFICATION OF LES USING LITTLE JOE II	C C C C	C	C	C C C C C		C C	C C				SP	P	R		P							
	BP-23 (MISSION COMPLETE)	WSMR	HIGH-Q ABORT TEST, VERIFICATION OF LES, ELS, CANARD, ETC	C C C C	C	C	C C C C C		C C	C C				SP	P	R		P							
	BP-23A (MISSION COMPLETE)	WSMR	C/M FOR PAD ABORT	C C C	C		C C C C C		C C	C C				SP	P	R		P							
LAUNCH ENVIRONMENT VEHICLES	BP-13 (MISSION COMPLETE)	CAPE KENNEDY	FIRST VEHICLE TO QUALIFY SATURN I	C C C	C C	C	S S C C		C C				S	SP	R & P	R		P							
	BP-15 (MISSION COMPLETE)	CAPE KENNEDY	SECOND VEHICLE TO QUALIFY SATURN I	C C C	C	C	C C C C		C C				S	SP	R & P	R		P							
DYNAMIC TEST VEHICLES	BP-9	MSFC	DETERMINATION OF DYNAMIC STRUCTURAL COMPATIBILITY WITH SATURN I LAUNCH VEHICLE	C C C	C		S S S S		P P P																
	BP-27	MSFC	DETERMINATION OF DYNAMIC STRUCTURAL COMPATIBILITY WITH SATURN IB, AND V LAUNCH VEHICLES	C C C	C		S S S S		P P P P	S S S	S	S	S S	S S	S S	S	S	S	S		S	S	S		
MICROMETEOROID EXPERIMENT VEHICLES	BP-16 (MISSION COMPLETE)	CAPE KENNEDY	EARTH ORBITAL MICRO-METEOROID EXPERIMENT	C C C	C	C C	S S C C		C C				S		P										
	BP-26 (MISSION COMPLETE)	CAPE KENNEDY	SAME AS BP-16	C C C	C	C	S S C C		C C				S		P										
	BP-9A (MISSION COMPLETE)	CAPE KENNEDY	SAME AS BP 16	C C C	C C	C	S S C C		C C				S		P										
UNMANNED FLIGHT VEHICLE	BP-30	CAPE KENNEDY	S/C & LEM PROPULSION TESTS	C C C C	(SLA)	C	S C		C C					P	R & P	R									C

LES · LITTLE JOE II · S-I

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Figure 7-2. Block I Boilerplate Vehicle Systems Configuration for Spacecraft Development

SYSTEMS CONFIGURATION LEGEND

- C - COMPLETE
- P - PARTIAL
- R - R & D INSTRUMENTATION EQUIPMENT ONLY
- S - SIMULATED OR INERT
- SP - SPECIAL
- M1 - CONTROL PROGRAMER
- M2 & M3 - APOLLO MISSION PROGRAMER FOR MANNED OR UNMANNED MISSIONS

NOTE:
THE INFORMATION IN THIS CHART IS BASED ON THE LATEST PROGRAM PLAN. NO ATTEMPT IS MADE TO SHOW IMPENDING CHANGES. THE CHARTS WILL BE UPDATED TO REFLECT OFFICIAL PROGRAM CHANGES DURING SUBSEQUENT REVISIONS OF THIS FAMILIARIZATION MANUAL.

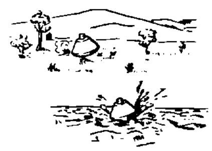
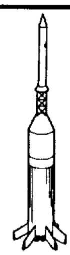
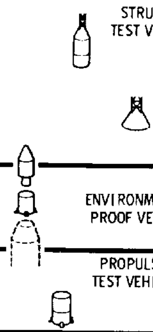
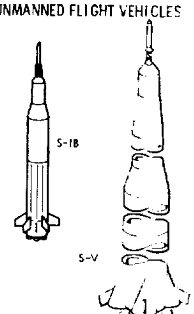
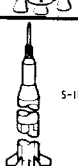
	SPACECRAFT	TEST SITES	MISSIONS OR PURPOSE	STRUCTURE (Tower, Command Module, Service Module, Boost Protective Cover)	SPACECRAFT ADAPTER (SLA, Extension, Special)	LAUNCH VEHICLES (Saturn I, Saturn IB, Saturn V, Little Joe II)	LAUNCH ESCAPE SYSTEM (LES) (Launch Escape Motor, Pitch Control Motor, Jettison Motor, Abort-Sequencer, Canard)	SERVICE PROPULSION SYSTEM (SPS)	SEPARATION SYSTEM (LES - CM, CM - SM, SM - Adapter, LEM - SLA)	EARTH LANDING SYSTEM (ELS) (Parachute, Recovery Aids, Recovery Sequencer, Flotation Equip.)	GUIDANCE & NAVIGATION SYSTEM (G&N)	STABILIZATION & CONTROL SYSTEM (SCS)	REACTION CONTROL SYSTEM (RCS) (Command Module, Service Module)	ENVIRONMENTAL CONTROL SYSTEM (ECS) (CM, SM)	ELECTRICAL POWER SYSTEM (EPS) (CM, SM)	COMMUNICATIONS SYSTEM	DEEP SPACE (S-BAND) EQUIP.	INSTRUMENTATION SYSTEM	CONTROLS AND DISPLAYS	APOLLO MISSION PROGRAMER	CREW COUCHES	SUPPORT & RESTRAINT	CREW SYSTEM	ANTHROPOMORPHIC DUMMIES	LUNAR EXCURSION MODULE (LEM)
EARTH RECOVERY AND IMPACT TEST VEHICLES 	S/C 002A	DOWNEY, CALIF.	S/C 002 REFURBISHED FOR LAND IMPACT TESTS	C														P			C	C		C	
	S/C 007	DOWNEY, CALIF. AND MSC	SPACECRAFT FOR ACOUSTIC, WATER IMPACT AND POST LANDING TESTS	SP C C S				S	S S S	S S S C P C C	S	S	S S	S S P	S S P	S P	S	P P	S P		S C	S C	S P	S C	
ABORT TEST VEHICLES  LITTLE JOE II	S/C 002	WSMR	POWER ON TUMBLING ABORT	C C C C	C	C	C C C C C		C C	C C P			S		P	P		P							
	S/C 010	WSMR	ABORT VEHICLE AND BACKUP FOR S/C 002	C C C C	C	C	C C C C C		C C	C P			S		P	P		P							
STRUCTURAL TEST VEHICLES 	S/C 004	DOWNEY, CALIF.	STATIC STRUCTURAL TESTS	SP C C SP	C			S	P P P	P															
	S/C 004A	DOWNEY, CALIF.	S/C 010 CM REFURBISHED FOR STATIC AND THERMAL STRUCTURAL TESTS	C					P																
	S/C 006	DOWNEY, CALIF.	C/M FOR ELS LOAD TESTS AND LET SEPARATION TESTS	C C					P																
ENVIRONMENTAL PROOF VEHICLE	S/C 008	MSC	EVALUATE S/C UNDER SIMULATED MISSION ENVIRONMENTAL CONDITIONS THERMAL VACUUM TESTS	C C	C			S	S S	C P C P	C	C	P C	C C	C C	C	P	C & SP	C		C	C	C		
PROPULSION TEST VEHICLES	S/C 001	(PSDF) WSMR	COMPATIBILITY VERIFICATION OF S/C SERVICE MODULE PROPULSION SYSTEMS WHEN INTEGRATED WITH S/M SUBSYSTEM.	C	C			C	P			SP	C	P	C			P							
UNMANNED FLIGHT VEHICLES  S-IB S-V	S/C 009	CAPE KENNEDY	SUPER CIRCULAR ENTRY FLIGHT TO QUALIFY S/C PRIOR TO MANNED FLIGHT	C C C C	C	C	C C C C	C	C C C	C C C C		P	C C	P P	P P	P		P	P	M1					
	S/C 011	CAPE KENNEDY	SUPER CIRCULAR ENTRY FLIGHT TO DEMONSTRATE OPERATION AND PERFORMANCE OF S/C AND SYSTEMS	C C C C	C	C	C C C C	C	C C C	C C C C	C	P	C C	C C	C C	C	P	C	C	M3					
	S/C 017	CAPE KENNEDY	SIMULATED LUNAR RETURN ENTRY	C C C C	C C	C	C C C C	C	C C C C	C C C C	P	P	C C	C C	C C	P	P	P	P	M3					P
	S/C 020	CAPE KENNEDY	SIMULATED LUNAR RETURN ENTRY	C C C C	C C	C	C C C C	C	C C C C	C C C C	P	P	C C	C C	C C	P	P	P	P	M3					P
MANNED FLIGHT VEHICLES  S-IB	S/C 012	CAPE KENNEDY	SEVEN-DAY EARTH ORBITAL FLIGHT & SEA RECOVERY TO EVALUATE IN-FLIGHT OPERATIONS	C C C C	C	C	C C C C	C	C C C	C C C C	C	C	C C	C C	C C	C	P	C	C	M2	C	C	C		
	S/C 014	CAPE KENNEDY	TEN-DAY EARTH ORBITAL FLIGHT TO EVALUATE IN-FLIGHT CSM OPERATIONS	C C C C	C	C	C C C C	C	C C C	C C C C	C	C	C C	C C	C C	C	P	C	C	M2	C	C	C		

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Figure 7-3. Block I Spacecraft Vehicle Systems Configuration for Spacecraft Development

SYSTEMS CONFIGURATION LEGEND

- C - COMPLETE
- P - PARTIAL
- S - SIMULATED OR INERT
- SP - SPECIAL
- M3 - APOLLO MISSION PROGRAMER FOR MANNED OR UNMANNED MISSIONS

NOTE: THE INFORMATION IN THIS CHART IS BASED ON PRELIMINARY DATA. NO ATTEMPT IS MADE TO SHOW IMPENDING CHANGES. THIS CHART WILL BE UPDATED TO REFLECT OFFICIAL PROGRAM CHANGES DURING SUBSEQUENT REVISIONS OF THIS FAMILIARIZATION MANUAL.

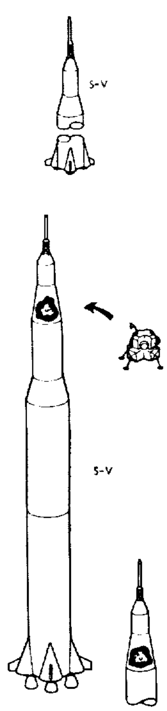
	SPACECRAFT NUMBER	TEST SITES	MISSIONS OR PURPOSE	STRUCTURE (Tower & Boost Protective Cover, Command Module, Service Module)	SPACECRAFT ADAPTER (SLA, Special)	LAUNCH VEHICLE (Saturn IB, Saturn V)	LAUNCH ESCAPE SYSTEM (Launch Escape Motor, Pitch Control Motor, Jettison Motor, Canard)	SERVICE PROPULSION SYSTEM (SPS)	SEPARATION SYSTEM (LES - CM, CM - SM, SM - Adapter, LEM - SLA)	SEQUENCE CONTROLLERS (Master Event, Earth Landing, SM-Separation, CM-RCS, Lunar Docking Events, LEM-SLA Separation)	RECOVERY SYSTEMS (ELS) (Earth Landing, Recovery Aids, Flotation Equip.)	GUIDANCE & NAVIGATION SYSTEM (G&N)	STABILIZATION & CONTROL SYSTEM (SCS)	REACTION CONTROL SYSTEM (RCS) (Command Module, Service Module)	ENVIRONMENTAL CONTROL SYSTEM (ECS) (CM, SM)	ELECTRICAL POWER SYSTEM (EPS) (CM, SM)	COMMUNICATIONS SYSTEM (Deep Space, Near Earth, Recovery)	INSTRUMENTATION SYSTEM (Operational, R & D, Scientific)	CONTROLS AND DISPLAYS	APOLLO MISSION PROGRAMMER	EMERGENCY DETECTION SYSTEM	CREW EQUIPMENT (Couches And Restraints, Accessories & Provisions, Anthropomorphic Dummies)	LUNAR EXCURSION MODULE (LEM)	LEM DOCKING SYSTEM
MAJOR GROUND TEST VEHICLES 	2H-1	DOWNEY, CALIF.	HOUSE SPACECRAFT NO. 3	P C P	C			S	P P P	C C C C C C	P P P	C	C	S S	P	C P	C C	P	C		P	C		C
RECOVERY TEST VEHICLES 	2S-1	DOWNEY, CALIF.	MODAL, ACOUSTIC, FLOTATION AND IMPACT	C C	C			C	S S S	S S S S S S	S S C	S	S	S C	S S	S S	S S	P	S			C S C	S	P
	007A	DOWNEY, CALIF.	S/C 007 REFURBISHED FOR BLOCK II POST LANDING TESTS	C						S S S S S	P C C	S	S	S	P	P	S C	P	P			C P C		
STATIC TESTS	2S-2	DOWNEY, CALIF.	S/M FOR STATIC TESTS	SP & P C C	C				P P P									P	S					P
ENVIRONMENTAL PROOF VEHICLE	2TV-1	MSC	THERMAL VACUUM TESTS	C C	C			S	S S S S	C C C C C C	C C C	C	C	S S	C C	C C	S S	C P P	C		C	C C C	C	C
MANNED FLIGHTS 	101	CAPE KENNEDY	ELLIPTICAL EARTH ORBITAL FLIGHT TO EVALUATE RENDEZVOUS AND DOCKING MANEUVERS	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C C	C P P	C		C	C C	C	C
	102	CAPE KENNEDY	SMALL ELLIPTICAL EARTH ORBITAL FLIGHT TO EVALUATE LEM AND CSM RENDEZVOUS AND DOCKING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C C	C P	C		C	C C	C	C
	103	CAPE KENNEDY	LUNAR LANDING RESEARCH AND DEVELOPMENT	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C C	C P	C		C	C C	C	C
	104	CAPE KENNEDY	LUNAR LANDING TO EVALUATE MAN ON LUNAR SURFACE AND DEMONSTRATE LEM CAPABILITY	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	105	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	106	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	107	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	108	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	109	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	110	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	111	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C
	112	CAPE KENNEDY	LUNAR LANDING	C C C	C	C	C C C C	C	C C C C	C C C C C C	C C C	C	C	C C	C C	C C	C C	C P	C		C	C C	C	C

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Figure 7-4. Block II Spacecraft Vehicle Systems Configuration for Spacecraft Development

7-8. Boilerplate and spacecraft vehicle systems configuration for spacecraft development of Block I and Block II vehicles is shown in figures 7-2 through 7-4. Letters are used to designate the following: a complete system (C), a partial system (P), R&D instrumentation equipment only (R), a simulated or inert system (S), a special system (SP), and different configurations of the Apollo mission programer (M1, M2, or M3). A blank space in any column indicates the described system is not installed.

7-9. BLOCKS I AND II.

7-10. A block concept is used for spacecraft development to separate the vehicles into different phases, such as research and development (Block I) and production vehicles for earth orbital and lunar missions (Block II). Paragraphs 7-11 through 7-14 give a breakdown of Blocks I and II and their functions.

NOTE

Block II information is based on preliminary data only.

7-11. Block I encompasses the entire boilerplate program, and spacecraft 001, 002, 002A, 004, 004A, 006, 007, 008, 009, 010, 011, 012, 014, 017, and 020.

7-12. The boilerplate portion of Block I provides:

a. Early support of systems development for land impact, water impact, and parachute recovery prequalification tests

b. Systems qualification to support spacecraft programs including pad abort, high-altitude abort, and house spacecraft No. 1 (boilerplate 14) which contains all systems

c. Marshall Space Flight Center support including Saturn I development and micrometeoroid detection

d. Space-flight capabilities development and coordination of manufacturing, testing operations, engineering, and NASA functions.

7-13. The spacecraft portion of Block I provides:

a. Command module and service module development for manned earth orbital missions

b. Demonstration of operational capabilities of systems including all types of aborts, land recovery, water recovery, Saturn IB and V operation (and compatibility), and operation during earth orbits (unmanned)

c. Qualified teams development for checkout, launch, manned space flight network, recovery, and flight analysis.

NOTE

S/C 007 will be refurbished and designated 007A for Block II postlanding tests.

7-14. Block II encompasses spacecraft 2H-1, 2S-1, 007A, 2S-2, 2TV-1, 101, 102, 103, 104, 105, 106, 107, 108, 109, 110, 111, and 112, and provides:

a. Incorporation of lunar excursion module provisions

b. Reduction of weight in command module

c. Improvement of center-of-gravity in command module

d. Evaluation and incorporation of system changes with respect to lunar mission and reliability impact

7-15. The primary differences between the systems of Block I and Block II manned spacecraft are listed as follows:

Spacecraft System	Block I	Block II
ELS	Nylon main parachute risers.	Steel cable main parachute risers.
ECS		Redundant coolant loop added.
		LEM pressurization controls added to C/M.
		Inverter added to pressure suit compressor for emergency purposes.
		Glycol reservoir quantity measurement system added.
EPS	S/M separation batteries A and B, used to separate C/M-S/M.	No S/M separation batteries—capabilities for C/M-S/M separation added to fuel cells.
		One flight bus (d-c) added.
		No frequency meter on MDC.
RCS	No fuel dump capability in C/M.	Rapid fuel dump C/M-RCS.
G&N	Interfaces with SCS.	Completely independent of SCS.
	Long-relief-eyepieces installed by astronauts.	Long-relief-eyepieces incorporated into sextant and telescope.
	CDUs electro-mechanical.	CDUs all electronic.
		AGC size reduced and memory capacity increased.
		Reduced size and weight of IMU.
		Redesigned navigation base.
SCS	Interfaces with G&N.	Completely independent of G&N.
	One FDAI	Two FDAIs
	AGCU	Gyro display coupler (GDC) in place of AGCU.
	Rate gyro assembly	BMAG assembly in place of RGA.
Crew system	No portable life support system.	Portable life support system (PLSS).
	Three one-man liferafts.	One three-man liferaft.
	Incandescent and fluorescent illumination.	Electroluminescent, fluorescent, and incandescent illumination.
	No EVT capabilities.	Extravehicular transfer (EVT) and tethering capabilities.
		Redesigned pressure suit.

Spacecraft System	Block I	Block II
T/C	C-band transponder.	C-band transponder in S/C 101 only.
	One S-band transponder.	Two S-band transponders.
	One S-band power amplifier.	Two S-band power amplifiers.
	No high-gain antenna.	High-gain antenna and controls.
		Additional VHF/AM capabilities.
	No rendezvous transponder and antenna.	Rendezvous radar transponder and antenna.
Docking system		Installed in Block II S/C only.

7-16. BOILERPLATE MISSIONS.

7-17. The boilerplate missions are primarily research and development tests to evaluate the structural integrity of the spacecraft and confirm basic engineering concepts relative to system performance and compatibility. A number of missions will be conducted during this phase of the test program. The missions are scheduled to follow a pattern of development starting with basic structure evaluation, followed by systems performance and compatibility confirmation.

7-18. Each mission is dependent upon the previous mission in developing the systems and operations requisite for lunar exploration. Prior to the start of any mission, the boilerplate to be tested is thoroughly checked at the manufacturer test preparation area under the direction of NASA inspectors. After system and structural checkout has been approved, the boilerplate is shipped to the test site for further checkout and mating. A launch countdown is started only after the second checkout and mating have been approved. Figure 7-5 depicts a water impact test.

7-19. BLOCK I BOILERPLATE TEST PROGRAM. The following is a list of each boilerplate and relative mission data. Boilerplates and their missions are part of the Block I portion of the Apollo program. The list is intended as a cross-reference for boilerplate objectives. Chronological order of missions and test grouping is not intended or reflected in the arrangement of the list.

Boiler-plate No.	Test Site	Purpose	Mission	Launch Vehicle
BP1	Downey, Calif.	Development and evaluation of crew shock absorption system; evaluation of C/M on land and water, during and after impact.	Drop tests utilizing impact facility at Downey, Calif.	None
BP2	Downey, Calif.	Same as BP1 and development of the uprighting system.	Drop tests and uprighting tests utilizing impact facility at Downey, Calif.	None

Block I Boilerplate Test Program (Cont)

Boiler-plate No.	Test Site	Purpose	Mission	Launch Vehicle
BP3	El Centro, Calif.	To evaluate parachute recovery system in the air (destroyed).	Parachute recovery via air drop.	Aircraft (drop)
BP6	White Sands Missile Range (WSMR), New Mexico	To determine aerodynamic stability, tower vibration, and spacecraft dynamics during a pad abort; to demonstrate capability of LES to propel C/M to a safe distance from launch area during a pad abort.	Pad abort mission successfully completed 7 November 1963.	Launch escape system motor
BP6A	El Centro, Calif.	BP6 refurbished for parachute recovery system test in the air.	Parachute recovery system evaluation via air drop.	Aircraft (drop)
BP6B	El Centro, Calif.	BP6A refurbished for parachute recovery system tests.	Parachute recovery system evaluation tests.	Aircraft (drop)
BP9	Marshall Space Flight Center (MSFC), Alabama	Dynamic test to be determined by National Aeronautics and Space Administration.	Determination of dynamic structural compatibility of test S/C with Saturn I.	None
BP9A	Kennedy Space Center, Florida	Utilized for launch vehicle qualification and micrometeoroid experiment.	Micrometeoroid experiment. Successfully launched into orbit 30 July 1965.	Saturn I
BP12	WSMR, New Mexico	To determine aerodynamic stability characteristics of Apollo escape configuration during transonic abort from Little Joe II. To demonstrate capability of LES to propel C/M to safe distance from launch vehicle during an abort at high-dynamic pressure. To demonstrate structural integrity of launch escape tower and operational characteristics during a transonic abort, and demonstrate spacecraft-Little Joe II compatibility.	Transonic abort mission successfully completed 13 May 1964.	Little Joe II

Block I Boilerplate Test Program (Cont)

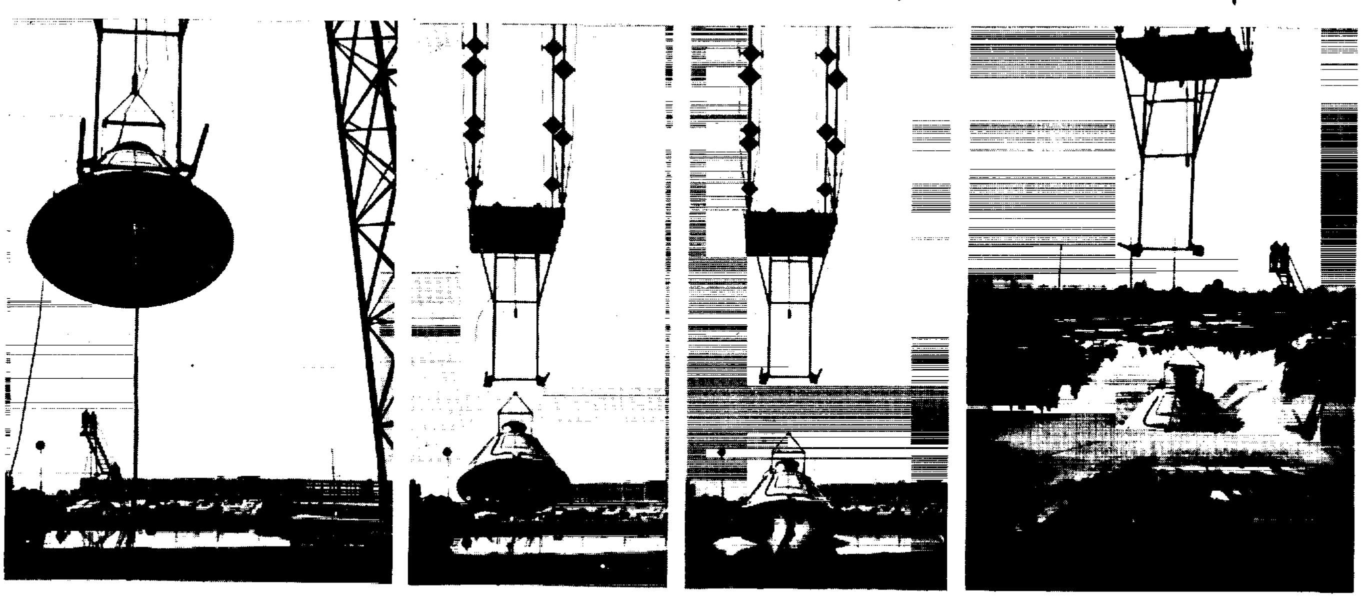
Boiler-plate No.	Test Site	Purpose	Mission	Launch Vehicle
BP12A	Downey, Calif.	BP12 refurbished to evaluate C/M during hard rollover water landing condition.	Water impact test utilizing impact facility at Downey, Calif.	None
BP13	Kennedy Space Center, Florida	To qualify Saturn I launch vehicle, demonstrate physical compatibility of launch vehicle and spacecraft under preflight and flight conditions, and determine launch and exit environmental parameters to verify design criteria.	Launch environment mission successfully completed 28 May 1964 to 31 May 1964.	Saturn I
BP14	Downey, Calif.	Developmental tool (house spacecraft No. 1) for use in developing spacecraft systems and preliminary checks in integrated systems compatibility.	Research and developmental tool for systems evaluation (static vehicle).	None
BP15	Kennedy Space Center, Florida	Second boilerplate flown for environmental data. The purpose of this vehicle and BP13 are similar.	Launch exit environment (orbital flight trajectory) mission successfully completed 18 September 1964 to 22 September 1964.	Saturn I
BP16	Kennedy Space Center, Florida	To be utilized for launch vehicle qualification and micrometeoroid experiment.	Micrometeoroid experiment. Successfully launched into orbit 16 February 1965.	Saturn I
BP19	El Centro, Calif.	To evaluate parachute recovery system in the air.	Parachute recovery system evaluation through means of command module air drop.	Aircraft (drop)
BP22	WSMR, New Mexico	To verify LES, ELS, and canard system during high-altitude abort.	Qualify LES, ELS sequence timing during abort. Mission completed 19 May 1965.	Little Joe II

Block I Boilerplate Test Program (Cont)

Boiler-plate No.	Test Site	Purpose	Mission	Launch Vehicle
BP23	WSMR, New Mexico	Verification of LES and ELS during high-Q abort.	Demonstration of launch escape vehicle structural integrity and recovery of C/M following high-Q abort. Successfully completed mission 8 December 1964.	Little Joe II
BP23A	WSMR, New Mexico	To qualify LES, ELS sequence timing, canard system, dual drogue parachutes, and boost protective cover during pad abort test. BP23 C/M refurbished for pad abort test.	C/M for pad abort evaluation. Mission completed 29 June 1965.	Launch escape system motor.
BP25	Houston, Texas	To demonstrate pickup and handling techniques as required by National Aeronautics and Space Administration.	Demonstration of equipment and handling capability for command module at a site (simulated) recovery as a design for pickup equipment.	None
BP26	Kennedy Space Center, Florida	To be utilized for launch vehicle qualification and micrometeoroid experiment.	Micrometeoroid experiment using NASA-installed equipment. Successfully launched into orbit 25 May 1965.	Saturn I
BP27	MSFC, Alabama	Second dynamic test. Objectives for this ground test will be determined by National Aeronautics and Space Administration.	Determination of dynamic structural compatibility of test S/C with Saturn IB and Saturn V launch vehicles.	Saturn IB and Saturn V (captive test firing)

Block I Boilerplate Test Program (Cont)

Boiler-plate No.	Test Site	Purpose	Mission	Launch Vehicle
BP28	Downey, Calif.	Test vehicle will be impacted (land and water) a number of times in order to evaluate loads imposed on structure due to landing impact and acceleration, accelerations and onset rates imposed on crew by the crew couch attenuation system, stability, and dynamics of the vehicle.	Definition of landing problems by determination and evaluation of loads imposed on structure due to landing impact, and acceleration and onset rates imposed on crew by crew couch attenuation system.	None
BP29	MSC Houston, Texas	To determine flotation characteristics of command module and to qualify Block I uprighting system.	Full-scale flotation and recovery tests for simulated entry and abort conditions.	None
BP30	Kennedy Space Center, Florida	An unmanned flight to evaluate S/C and LEM propulsion performance.	Elliptical orbital flight for CSM and LEM operation.	Saturn IB



A PENDULUM FOR APOLLO—An impact test facility, from which NASA's unmanned Apollo test command modules are swung and dropped on land or water, is seen in operation at Downey, Calif. The Apollo command module is suspended below the steel platform and the huge "arm" swings the capsule, releasing it at controlled angles and speeds to simulate impact which later manned Apollo spacecraft will undergo upon return to earth.

SM-2A-488A

Figure 7-5. Structural Reliability Test

7-20. SPACECRAFT MISSIONS.

7-21. Spacecraft missions, conducted with production spacecraft, are of a multipurpose nature. The initial missions will be conducted to verify production spacecraft structural integrity, and systems operation and compatibility. After the spacecraft structure and systems test missions are completed, a series of unmanned missions will be conducted to confirm spacecraft-launch vehicle compatibility and evaluate prelaunch, launch, mission, and postmission operations. Manned spacecraft missions will be conducted to improve performance and confirm capability between man and spacecraft.

7-22. Each spacecraft mission is dependent upon the success of the previous mission, as an overall program of progressive, manned, space penetration is planned to afford the flight crew maximum familiarity with production spacecraft maneuvers and docking techniques with the LEM. Manned missions will penetrate deeper into space as the program progresses. Manned space-flight network techniques will be employed during all earth orbital missions. The system compatibility with Apollo spacecraft will be determined during this phase of the test program. The knowledge gained by the flight crew and engineering personnel during this phase will be analyzed for use during manned lunar surface exploration.

7-23. BLOCK I SPACECRAFT TEST PROGRAM. The following is a complete list of Apollo spacecraft, their missions, and relative data required to complete the Block I portion of the Apollo program. The list is similar in intent to that of paragraph 7-19; the chronological order of spacecraft missions is not intended or reflected in the arrangement of the list.

Spacecraft No.	Test Site	Purpose	Mission	Launch Vehicle
001	Propulsion System Development Facility (PSDF), White Sands, Missile Range (WSMR), New Mexico	To verify compatibility of spacecraft S/M propulsion system; evaluate S/M service propulsion system and reaction control system during malfunction, normal, and mission profile conditions. To evaluate interface compatibility between all onboard systems during integrated systems test; evaluate performance and compatibility pertaining to ground support equipment, safety rules, operating techniques, and checkout of applicable systems.	Determine spacecraft structural vibration and acoustic characteristics during all phases of SPS operation, and performance of electrical power system and cryogenic storage subsystem during service propulsion and reaction control systems operation.	None (See figure 7-6.)
002	WSMR, New Mexico	To demonstrate structural integrity of production C/M under high dynamic pressure at transonic speed. To determine operational characteristics during power on tumbling abort.	Abort at high dynamic pressures in transonic speed range.	Little Joe II

Block I Spacecraft Test Program (Cont)

Spacecraft No.	Test Site	Purpose	Mission	Launch Vehicle
002A	Downey, Calif.	S/C 002 refurbished for land impact tests to verify structural integrity of C/M and assure acceptable crew accelerations during land impact.	Land impact tests utilizing impact facility at Downey, Calif.	None
004	Downey, Calif.	S/C for verification of structural integrity and intramodular compatibility of combined module structures under critical loadings.	Static tests.	None
004A	Downey, Calif.	Verification of structural integrity and intramodular compatibility of combined module structures under critical loading.	Static and thermal structural tests.	None
006	Downey, Calif.	C/M ELS load tests and C/M-LET separation tests.	Systems evaluation.	None
007	Downey, Calif. and MSC	This spacecraft will serve a dual purpose: module transmissibility and water impact and flotation tests. Module transmissibility test will determine free-fall lateral bending modes, and longitudinal and shell modes, utilizing two spacecraft configurations. Configuration A will incorporate launch escape tower and C/M; configuration B will incorporate C/M and S/M. Water impact and flotation tests will utilize C/M only. Purpose of water impact test is to verify structural integrity of C/M and crew support system dynamics under shock conditions at water impact. Flotation tests will demonstrate C/M flotation and water-tight integrity in varying sea conditions as well as crew survival in a closed module, and crew egress in varying sea conditions.	Acoustic, water impact, and post-landing tests.	None
008	MSC, Houston, Texas and Downey, Calif.	Spacecraft 008 will undergo both manned and unmanned deep-space environmental control tests. The first of these tests will be conducted at Downey, Calif. After	Thermal vacuum tests.	None

Block I Spacecraft Test Program (Cont)

Spacecraft No.	Test Site	Purpose	Mission	Launch Vehicle
		operational checkout of installed systems, spacecraft will be shipped to MSC facility at Houston, Texas, for evaluation and verification of complete spacecraft design under launch simulation, orbital mission simulation, thermal investigation, system failure increments, emergency operations, module separation, entry separation, and recovery aids checkout.		
009	Kennedy Space Center, Florida	To evaluate heat shield ablator performance, RCS and SPS operations, SCS operation, open loop EDS and partial EPS operation. To determine loading separation characteristics and communications performance. To demonstrate operation of recovery system, launch vehicle and spacecraft compatibility, and cryogenic gas storage system operation. A mission programer, model M1, will control S/C in-flight operations.	Supercircular high-heat rate entry flight.	Saturn IB
010	WSMR, New Mexico	After static tests as S/C 004A, this C/M will undergo an abort test and will be used as a backup vehicle for S/C 002.	Qualification of launch escape configuration of production spacecraft.	Little Joe II
011	Kennedy Space Center, Florida	An unmanned flight to evaluate heat shield ablator performance, EDS performance, and SPS propellant retention device. Determine SLA structural loading, separation characteristics, performance of G&N, SCS, ECS, EPS, RCS, and telecommunications. To demonstrate launch vehicle and S/C compatibility, structural integrity, C/M entry, multiple SPS restarts, and recovery. S/C will be controlled by model M3 mission programer.	A supercircular high-heat load entry flight.	Saturn IB

Block I Spacecraft Test Program (Cont)

Spacecraft No.	Test Site	Purpose	Mission	Launch Vehicle
012	Kennedy Space Center, Florida	A manned flight to evaluate crew-S/C compatibility, crew tasks and subsystems performance, manual, and backup modes of subsystem separations. To demonstrate closed loop EDS.	A long-duration orbital flight for CSM subsystem performance.	Saturn IB
014	Kennedy Space Center, Florida	A manned flight to evaluate in-flight CSM performance. To determine radiation levels and to demonstrate closed loop lifting entry.	Elliptical orbital flight for CSM operation.	Saturn IB
017	Kennedy Space Center, Florida	An unmanned flight to evaluate mission programer performance, ECS entry performance, heat shield performance, countdown operations, Saturn V performance, MSFN ability, SCS entry performance, G&C during entry, and G&C boost monitor. To determine open-loop EDS performance, boost environment, structural loading, SPS performance, radiation levels ECS, and EPS operation. To demonstrate structural performance, G&C effectiveness, sea recovery compatibility, and parachute recovery.	Structural integrity and simulated lunar return high-heat rate entry.	Saturn V
020	Kennedy Space Center, Florida	An unmanned flight to evaluate heat shield performance, countdown, launch vehicle repeatability, MSFN ability, LEM propulsion performance, and LEM G&C stability. To determine open-loop EDS performance, ECS entry performance, LEM G&C control, and LEM subsystem performance. To demonstrate LEM performance, SPS performance, LEM fluid systems, LEM separation, and sea recovery.	Simulated lunar return high-heat load entry and LEM propulsion.	Saturn V

7-24. BLOCK II SPACECRAFT TEST PROGRAM. The following is a complete list of Apollo spacecraft, their missions, and relative preliminary data required for the Block II portion of the Apollo program.

Spacecraft No.	Test Site	Purpose	Mission	Launch Vehicle
2H-1	Downey, Calif.	Verification of systems performance between major modules and associated ground support equipment.	House test model.	None
2S-1	Downey, Calif.	Water and land impact tests to verify structural integrity of C/M and assure acceptable crew accelerations during land impact.	Water and land impact tests utilizing impact facility at Downey, Calif.	None
007A	Downey, Calif.	S/C 007 refurbished for Block II postlanding tests.	Recovery test vehicle.	None
2S-2	Downey, Calif.	To verify structural integrity of CSM.	CSM for static structural tests.	None
2TV-1	MSC, Houston, Texas	To evaluate S/C under simulated mission environmental conditions.	Environmental proof (thermal vacuum) tests.	None
101	Kennedy Space Center, Florida	To evaluate rendezvous maneuvers, rendezvous radar transponder, guidance and control, and RCS plume effects. To determine LEM propulsion effects and heat shield performance. To demonstrate G&C entry, docking, LEM ECS operation, transposition and docking, LEM ΔV to CSM, landing gear deployment, crew transfer, one-man operation, and sea recovery.	Systems evaluation long-duration elliptical manned earth orbital flight.	Saturn IB
102	Kennedy Space Center, Florida	To evaluate SCS, LEM, and CSM in deep space. To determine LEM restart effects, SPS effectiveness, plume effects, and heat shield performance. To demonstrate CSM-LEM and manual entry.	Manned, small elliptical, long-duration earth orbital flight.	Saturn V

Block II Spacecraft Test Program (Cont)

Spacecraft No.	Test Site	Purpose	Mission	Launch Vehicle
103	Kennedy Space Center, Florida	Research and development to evaluate LEM and CSM operations, man on lunar surface, and to demonstrate LEM capability.	Manned lunar landing flight.	Saturn V
104	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
105	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
106	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
107	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
108	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
109	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
110	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
111	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V
112	Kennedy Space Center, Florida	Lunar landing.	Manned lunar landing flight.	Saturn V

7-25. TEST FIXTURES.

7-26. Three service propulsion engine test fixtures will be used to test the service module service propulsion systems. A test fixture (figure 7-6) is a structure used for system predevelopment and developmental tests leading to design of a spacecraft article. The fixtures are designated F-1, F-2, and F-3.

7-27. The F-1 test fixture functions to provide a test bed for the service propulsion engine vendor-acceptance, reliability, and qualification tests. The fixture will be used to evaluate the engine for safe operation and performance, to evaluate service propulsion engine basic design parameters, perform early evaluation of propellant system interaction effects, and to evaluate overall compatibility of the propulsion system components and subsystems.

7-28. The F-2 test fixture is a boilerplate structure that simulates the service module service propulsion system. This fixture will be used at PSDF, WSMR, New Mexico, to evaluate the service propulsion system under normal-design limit, and mission flight conditions through hot-propulsion static ground tests. The fixture will be used to permit continuance of the service propulsion system test program during periods when the propulsion spacecraft is out of service for modification, maintenance, and malfunction simulation.

7-29. The F-3 test fixture will be used at AEDC for vendor-acceptance and reliability tests. The F-3 fixture will also be used for engineering system development and static checkout tests by Space and Information Systems Division of North American Aviation, Inc.

7-30. GROUND SUPPORT EQUIPMENT.

7-31. Ground support equipment (GSE), required for Apollo spacecraft, is separated into four categories: checkout, auxiliary, servicing, and handling. The purpose of GSE is to provide the Apollo program with a GSE system that will establish a level of confidence in the onboard spacecraft systems and will ensure mission success within prescribed reliability factors.

7-32. CHECKOUT EQUIPMENT. Checkout equipment consists of acceptance checkout equipment (ACE), special test units (STU), bench maintenance equipment (BME), boilerplate and associate checkout equipment, and cabling systems.

7-33. Acceptance checkout equipment consists of permanently installed equipment located in control and computer rooms and carry-on (portable) equipment located near or onboard the spacecraft. Carry-on equipment is removed from the S/C prior to launch. ACE is computer-controlled equipment which provides the capability to check out spacecraft systems and to isolate malfunctions to a removable module. ACE also controls spacecraft systems servicing equipment.

7-34. Special test units provide the equipment to support the development of spacecraft systems and ACE systems. STU consists of manual checkout equipment and is required to operate and monitor the functional performance of the spacecraft systems.

7-35. Bench maintenance equipment is provided to perform verification and recertification, confirm defects, isolate malfunctions, perform repair verifications, perform calibration, and make adjustments on spacecraft systems, subsystems, and some components (to the lowest replaceable unit).

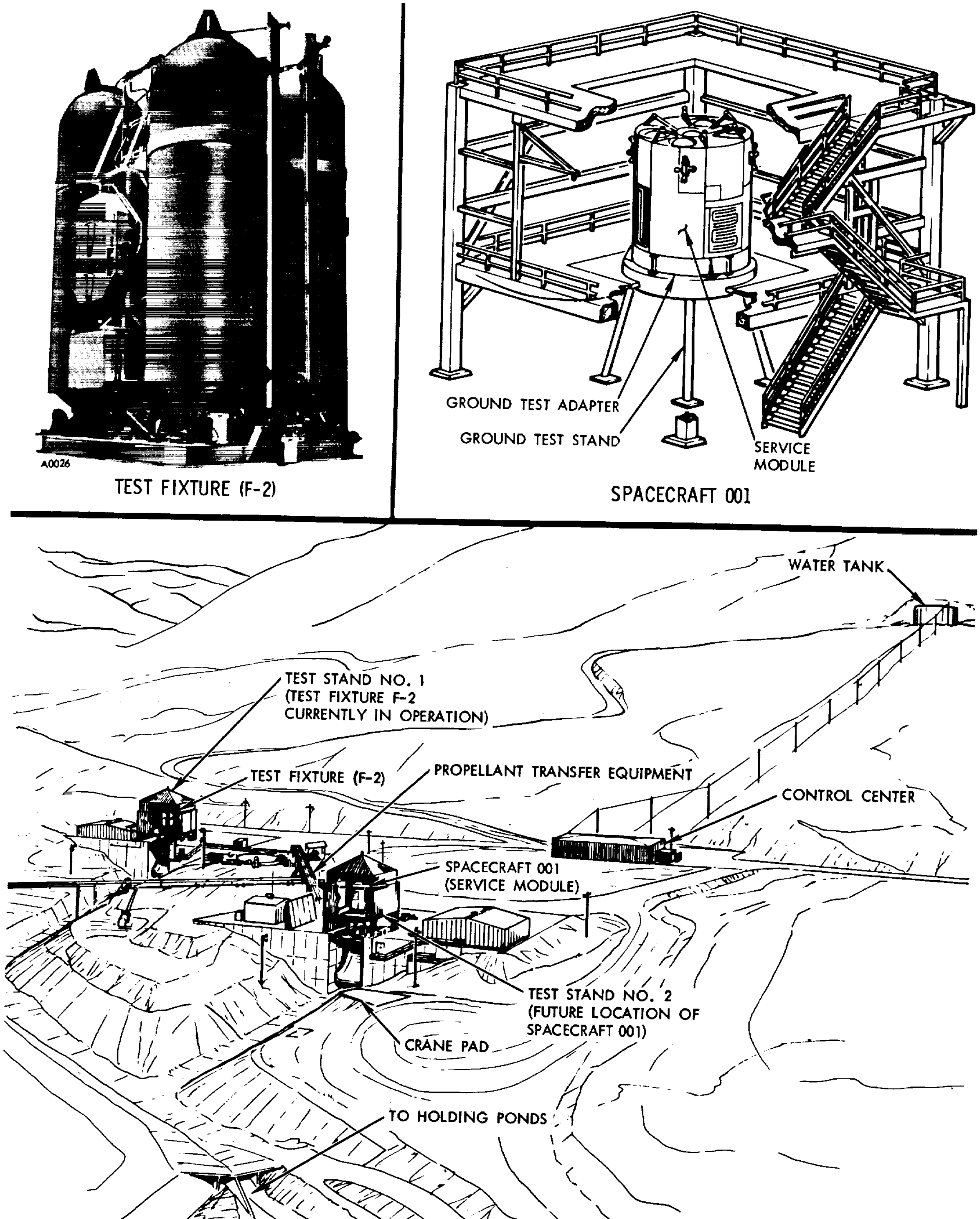


Figure 7-6. Test Fixture (F-2) and Spacecraft 001 at Test Site

7-36. Boilerplate and associate checkout equipment consists of equipment that cannot be classified as ACE, STU, or BME. Boilerplate checkout equipment, such as the Apollo R&D instrumentation console and the onboard record checkout unit, are used to check out some boilerplates. Associate checkout equipment, such as the R-F checkout unit, supports the spacecraft systems when ACE or STU checkout equipment is being used, and equipment that cannot be isolated to one particular spacecraft system, such as the mobile recorder and spacecraft ground power supply and power distribution panel.

7-37. Cabling systems include that equipment necessary to provide electrical interconnection between various spacecraft vehicles, ground equipment, and test facilities, as required to provide an integrated electrical checkout station.

7-38. AUXILIARY EQUIPMENT. Auxiliary equipment consists of accessory equipment and special devices, such as substitute units, alignment equipment, and protective closures, which are not a part of the checkout, servicing, and handling equipment.

7-39. Substitute units, such as the launch escape tower substitute unit, service module substitute unit, and command module substitute unit, provide the intermodule interface required to support an interface compatibility checkout of the electrically unmated command module or service module.

7-40. Alignment equipment, such as the optical alignment set and the command module optical alignment support, provide the test fixtures required to accomplish alignment tasks.

7-41. Protective closures, both hard and soft, provide covering for flight equipment during transportation and storage.

7-42. SERVICING EQUIPMENT. Servicing equipment consists of fluid handling equipment necessary to support the spacecraft systems during ground operation, and fluid handling equipment necessary to permit the onboard loading of all liquids and gases required to operate the spacecraft systems. Servicing equipment provides additional capabilities such as flushing, purging, conditioning, vapor disposal necessary to support the basic servicing function, and decontaminating the spacecraft systems as required after detanking.

7-43. HANDLING EQUIPMENT. Handling equipment provides for lifting, transportation, weight and balance, alignment, access, protection, and support of the service module, command module, and launch escape assembly.

7-44. MISSIONS COMPLETED.

7-45. The following missions have been successfully completed.

7-46. BOILERPLATE 6.

7-47. Boilerplate 6, an unmanned, pad abort test vehicle, using the launch escape and pitch control motors as a launch vehicle, successfully completed its mission at the White Sands Missile Range, New Mexico, 7 November 1963. (See figure 7-7.) An abort command caused the launch escape and pitch control motors to ignite, lifting the command module from the launch pad adapter. At an approximate altitude of 5000 feet, the launch escape assembly and forward heat shield were separated from the C/M, and the tower jettison motor ignited, propelling the launch escape assembly and forward heat shield clear of the C/M trajectory. The earth landing system was initiated to accomplish drogue parachute deployment, and release and deployment of three pilot parachutes which, in turn, deployed the three main parachutes, slowing the C/M to a safe landing speed (approximately 25 feet per second). Boilerplate 6 is being refurbished for further parachute recovery system tests and will be designated boilerplate 6A.

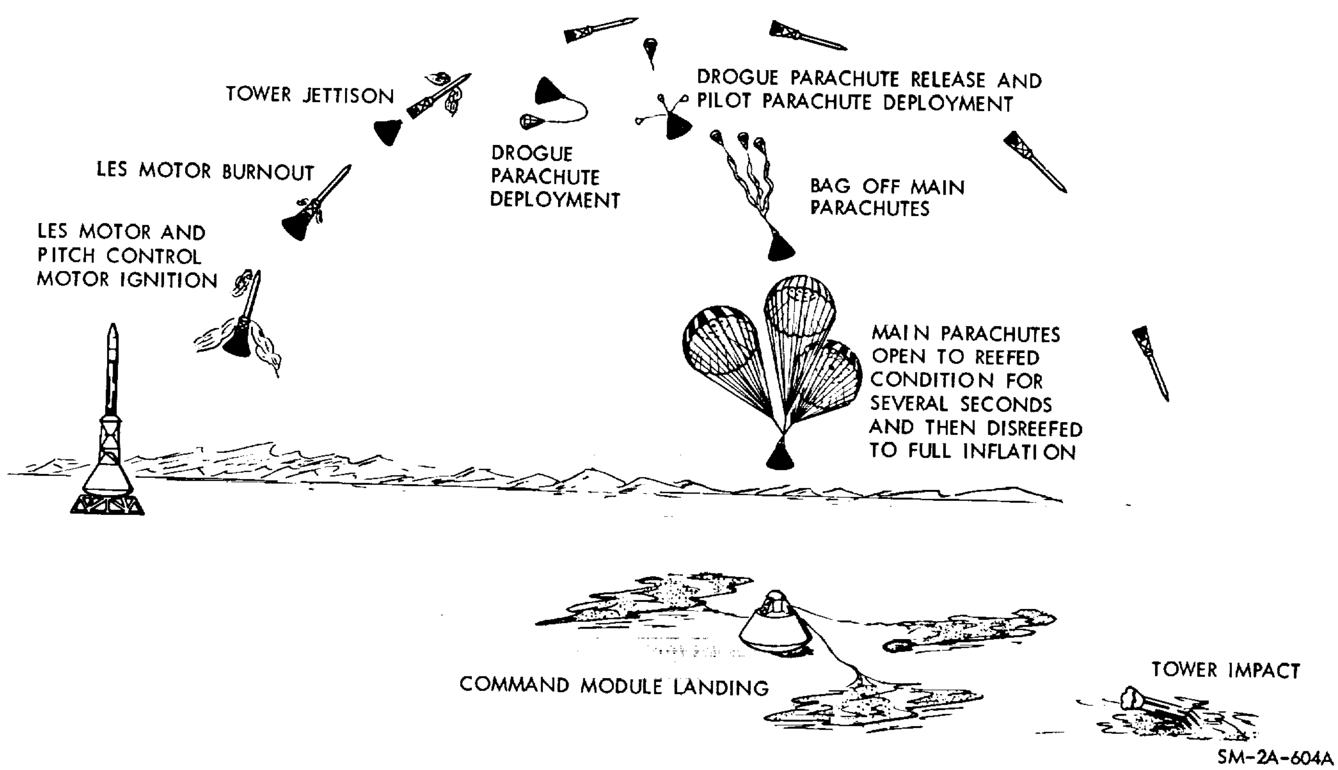


Figure 7-7. Boilerplate 6 Mission Profile

7-48. BOILERPLATE 12.

7-49. Boilerplate 12, an unmanned, transonic abort test vehicle, using a Little Joe II booster as a launch vehicle, successfully completed its mission at WSMR, New Mexico, 13 May 1964. (See figure 7-8.) This was the first full-scale test flight of the launch escape system in the transonic speed range. The Little Joe II boosted the boilerplate command-service modules to an approximate altitude of 21,000 feet, where an abort command caused separation of the C/M from the S/M and ignition of the launch escape and pitch control motors. The launch escape assembly propelled the command module away from the S/M and launch vehicle to an approximate altitude of 28,000 feet. The tower was separated from the C/M and the tower jettison motor ignited, carrying the launch escape assembly and forward compartment heat shield away from the trajectory of the C/M. The earth landing system was then initiated to accomplish drogue parachute deployment and release, and deployment of three pilot parachutes which, in turn, deployed the three main parachutes. One main parachute did not inflate fully and was separated from the C/M; however, the boilerplate 12 command module landed upright and undamaged.

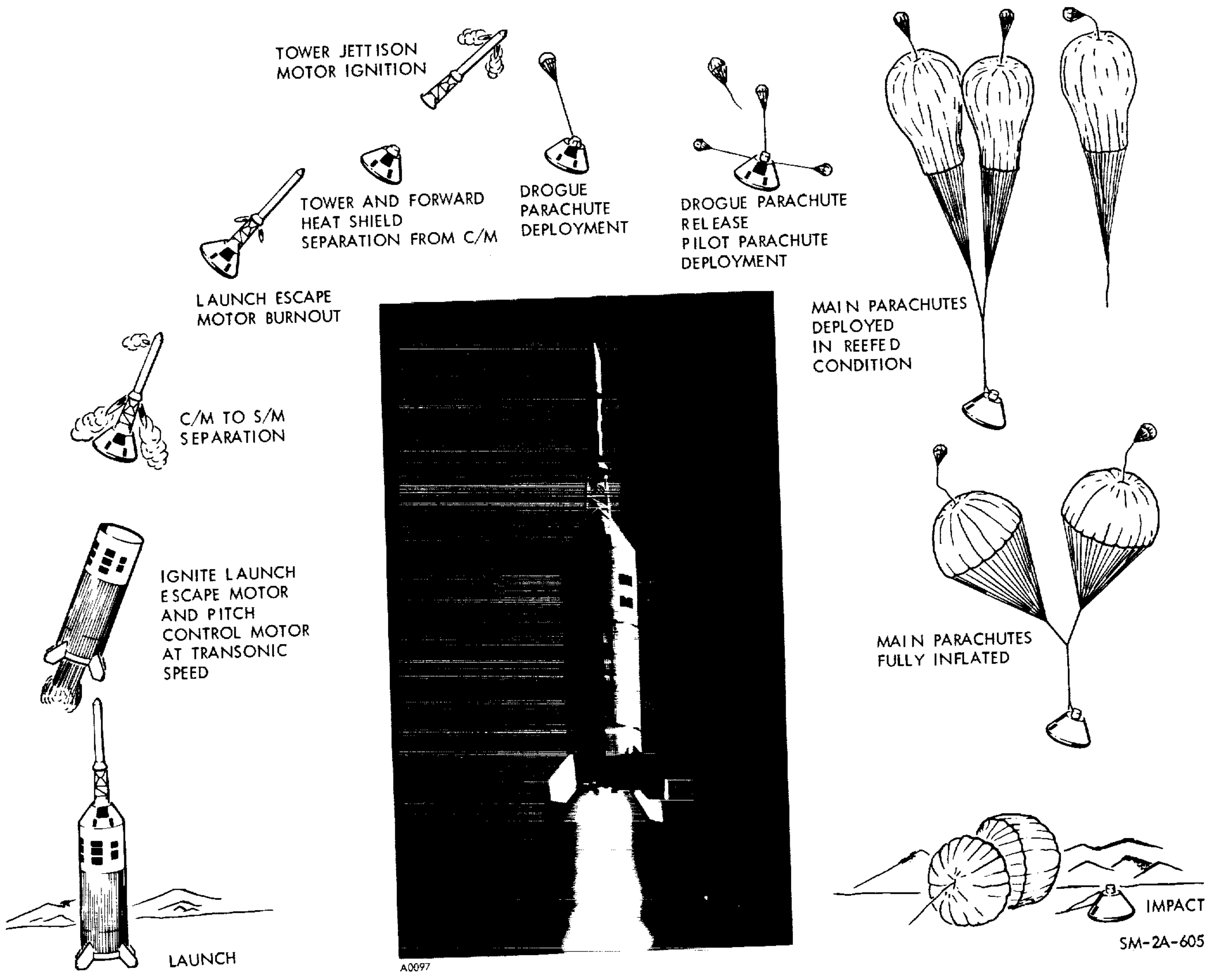


Figure 7-8. Boilerplate 12 Mission Profile

7-50. BOILERPLATE 13.

7-51. Boilerplate 13, an unmanned, launch environment test vehicle, using a Saturn I as a launch vehicle, was successfully launched into orbit from Kennedy Space Center, Florida, 28 May 1964. (See figure 7-9.) This was the first test flight to qualify the Saturn I launch vehicle and to demonstrate the compatibility of the spacecraft and launch vehicle. All test objectives were met; design parameters and conclusions about the flight, based on ground research, were as predicted. Orbits of the CSM and second-stage booster ranged from 110 to 140 miles above the earth surface and continued until 31 May 1964. Upon entry into earth atmosphere, the test vehicle disintegrated, as no provisions were made for recovery.

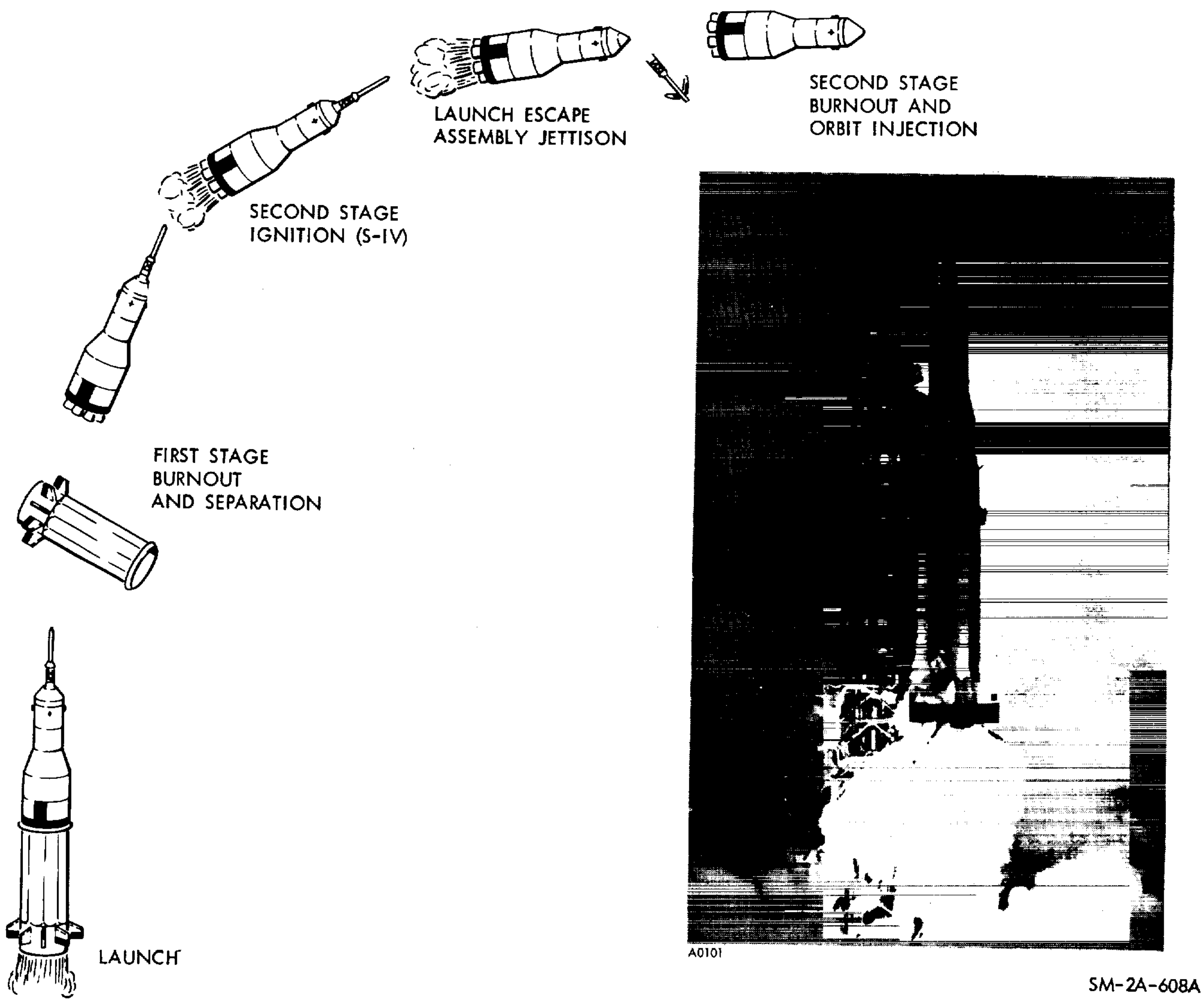


Figure 7-9. Boilerplate 13 Mission Profile

7-52. BOILERPLATE 15.

7-53. Boilerplate 15, an unmanned, launch environment test vehicle, using a Saturn I as a launch vehicle, was successfully launched into orbit from Kennedy Space Center, Florida, 18 September 1964. (See figure 7-10.) This was the second successful test flight to qualify the Saturn I launch vehicle and to demonstrate compatibility of the spacecraft and launch vehicle. An alternate mode of jettisoning the launch escape assembly was also demonstrated. Orbits of the CSM and second-stage booster ranged from 115 to 141 miles above the earth's surface and continued until 22 September 1964. No provisions were made for recovering the test vehicle upon entry into the atmosphere of the earth.

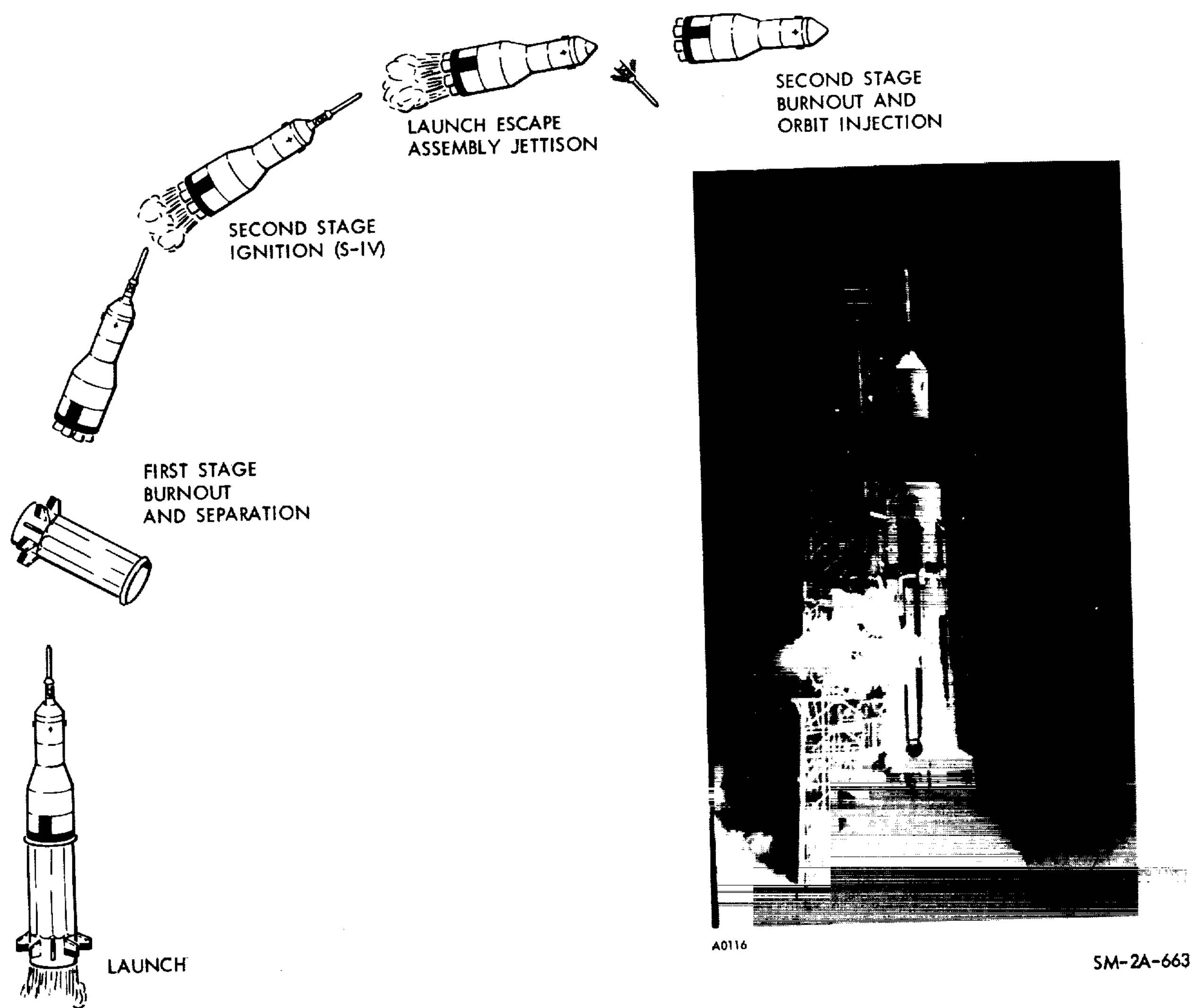


Figure 7-10. Boilerplate 15 Mission Profile

7-54. BOILERPLATE 23.

7-55. Boilerplate 23, an unmanned, abort test vehicle, using a Little Joe II booster as a launch vehicle, successfully completed its mission at WSMR, New Mexico, 8 December 1964. (See figure 7-11.) At approximately 32,000 feet a radio command signalled the launch vehicle control system to produce a pitch-up maneuver, simulating an abort condition. An abort command was initiated at an approximate altitude of 35,000 feet. Upon receipt of the abort signal, the C/M-S/M separated, and the launch escape and pitch control rocket motors ignited, to carry the C/M away from its launch vehicle. Eleven seconds after abort was initiated, the canards deployed, turning the C/M around and stabilizing it in a blunt-end forward attitude. At approximately 25,000 feet, the launch escape assembly separated from the C/M and the tower jettison motor ignited, carrying the launch escape assembly, boost protective cover, and forward heat shield away from the C/M. The earth landing system was initiated, accomplishing: drogue parachute (2) deployment in a reefed condition; drogue parachutes disreefed (at approximately 11,000 feet) slowing the speed and oscillation of the C/M for main parachute deployment, drogue parachute release, pilot parachute (3) deployment which, deployed the three main parachutes in a reefed condition; main parachutes disreefed lowering C/M to ground at a safe landing speed (approximately 25 feet per second). Boilerplate 23 is being refurbished for further C/M pad abort tests and will be designated boilerplate 23A.

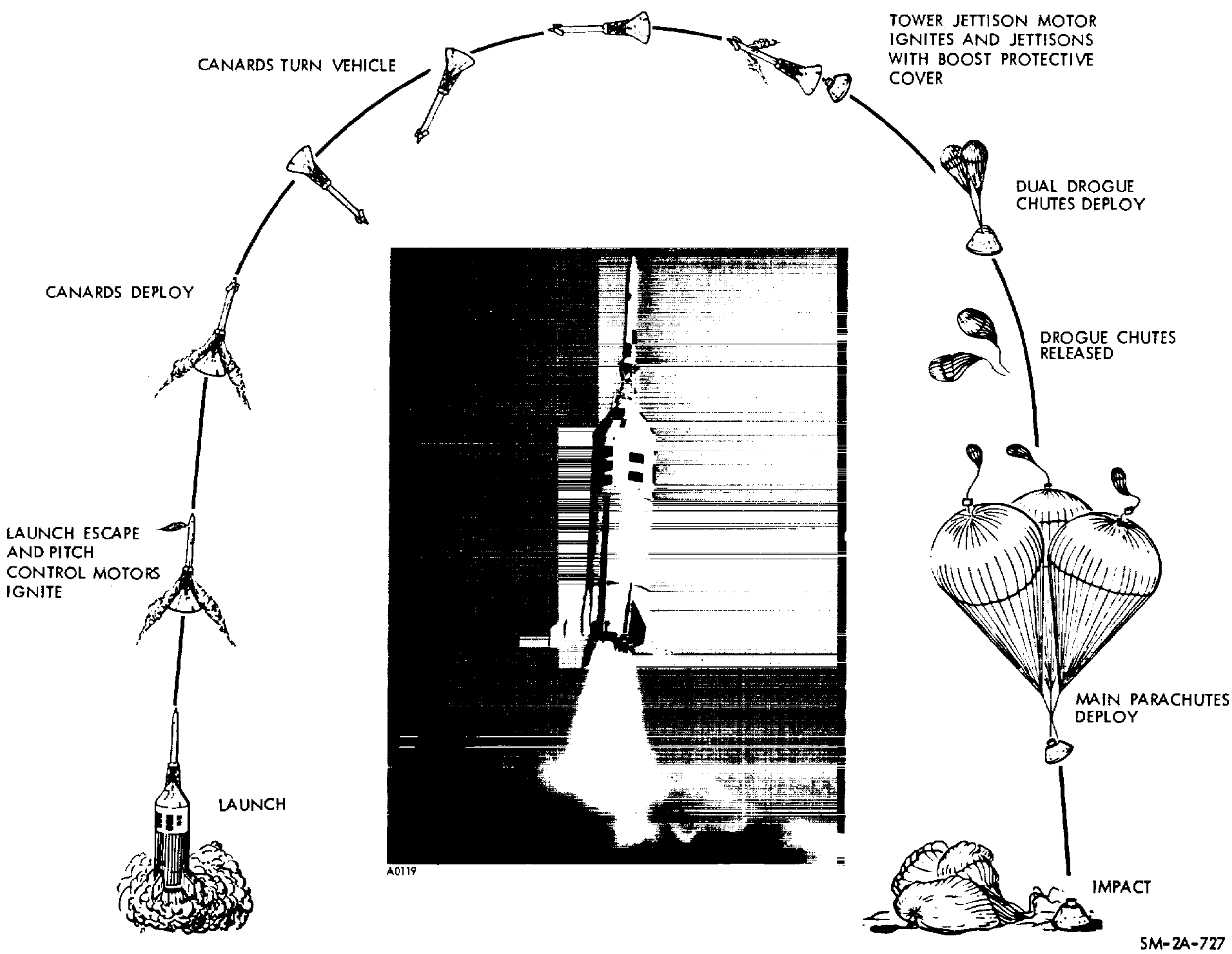


Figure 7-11. Boilerplate 23 Mission Profile

7-56. BOILERPLATE 16.

7-57. Boilerplate 16, an unmanned, micrometeoroid experiment test vehicle, using a Saturn I as a launch vehicle, was successfully launched into orbit from Kennedy Space Center, Florida, 16 February 1965. (See figure 7-12.) Once the orbit was attained, the CSM was jettisoned from the second stage (S-IV) by using the LES, and two large (NASA-installed) panels, unfolded. The panels and associated electronics, installed in the S-IV, are used to detect micrometeoroid particles and transmit the information to ground stations. The orbit of the test vehicle ranges from 308 to 462 miles above the earth. No provisions have been made for recovering the test vehicle upon entry into the atmosphere of the earth.

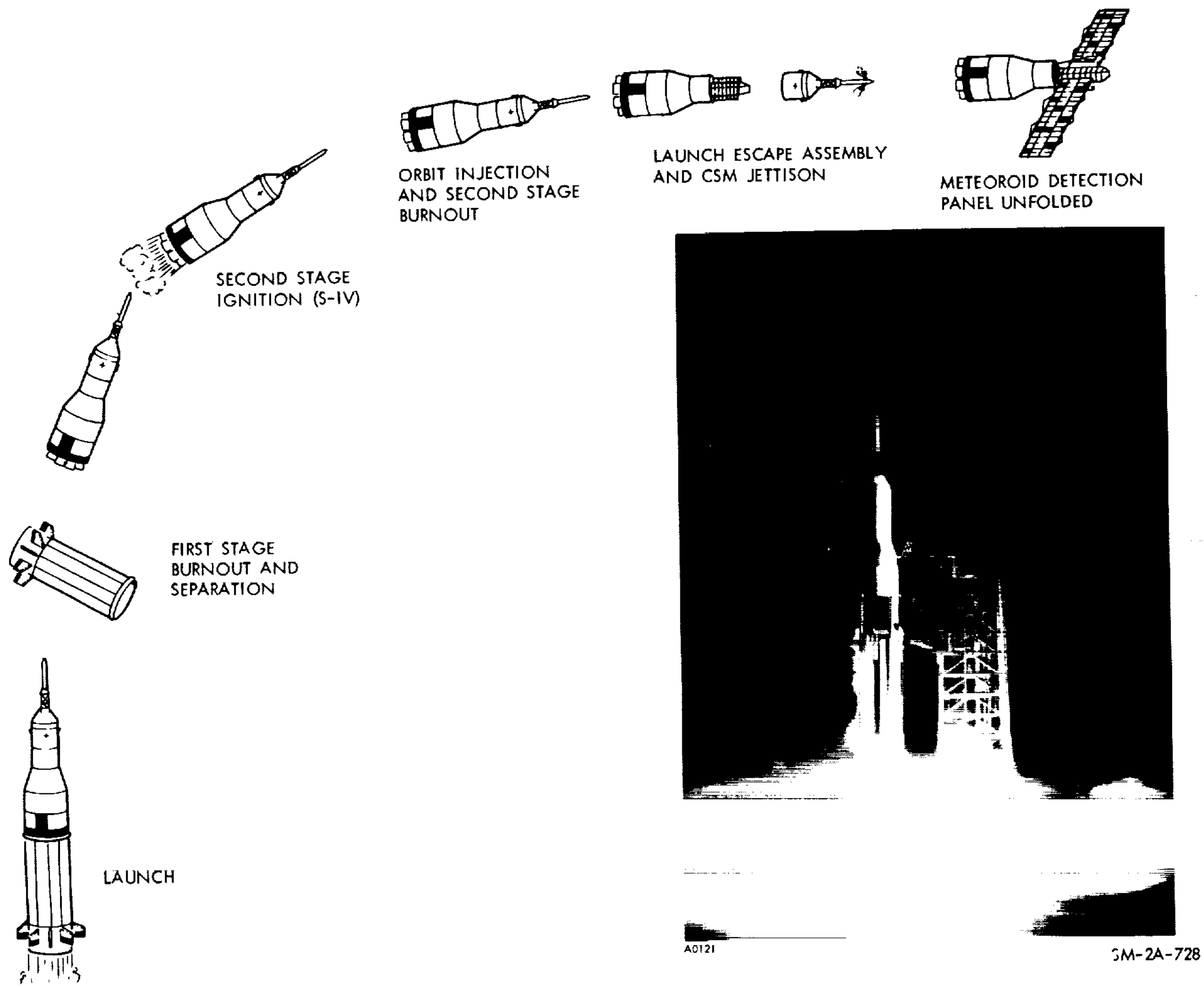


Figure 7-12. Boilerplate 16 Mission Profile

7-58. BOILERPLATE 22.

7-59. Boilerplate 22, an unmanned abort test vehicle using a Little Joe II booster as a launch vehicle, was partially successful in completing its mission at WSMR, New Mexico, 19 May 1965. (See figure 7-13). Although a high-altitude abort was planned, the boost vehicle malfunctioned causing a premature low-altitude abort; however, the Apollo systems functioned perfectly. An abort command was initiated due to the malfunctioning boost vehicle. Upon receipt of the abort signal, the C/M-S/M separated, and the launch escape and pitch control rocket motors ignited carrying the C/M away from the launch vehicle debris. The earth landing system was initiated lowering the C/M safely to the ground.

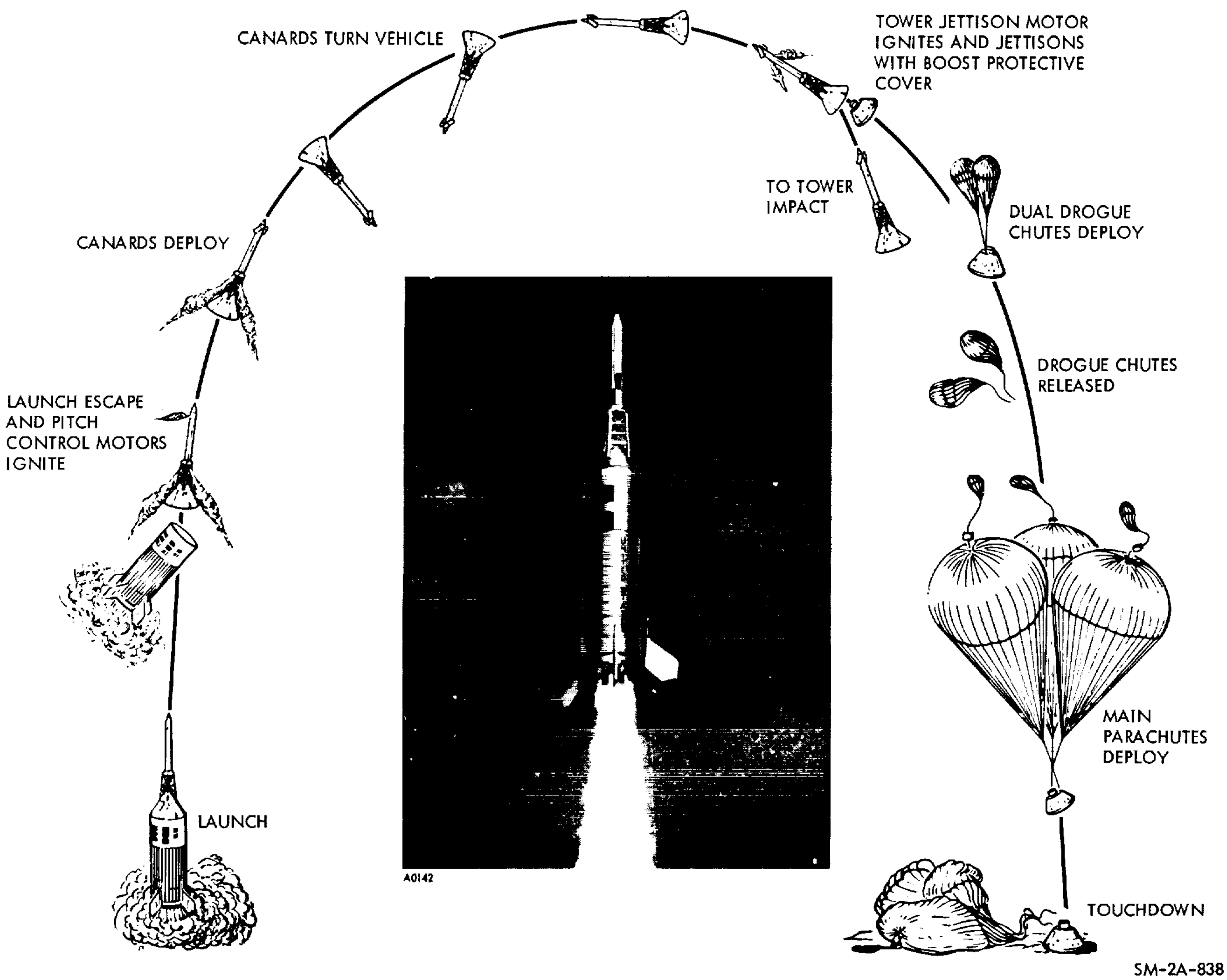


Figure 7-13. Boilerplate 22 Mission Profile

7-60. BOILERPLATE 26.

7-61. Boilerplate 26, the second unmanned micrometeoroid experiment test vehicle, was successfully launched into orbit from Kennedy Space Center, Florida, 25 May 1965. The launch vehicle used was a Saturn I. Using the LES, the CSM was jettisoned from the second stage (S-IV) upon reaching the planned orbit. (See figure 7-14.) Two large (NASA-installed) panels installed in the S-IV, unfolded, to be used with associated electronics to detect micrometeoroid particles and transmit the information to ground stations. No provisions have been made for recovering the test vehicle upon entry into the atmosphere of the earth.

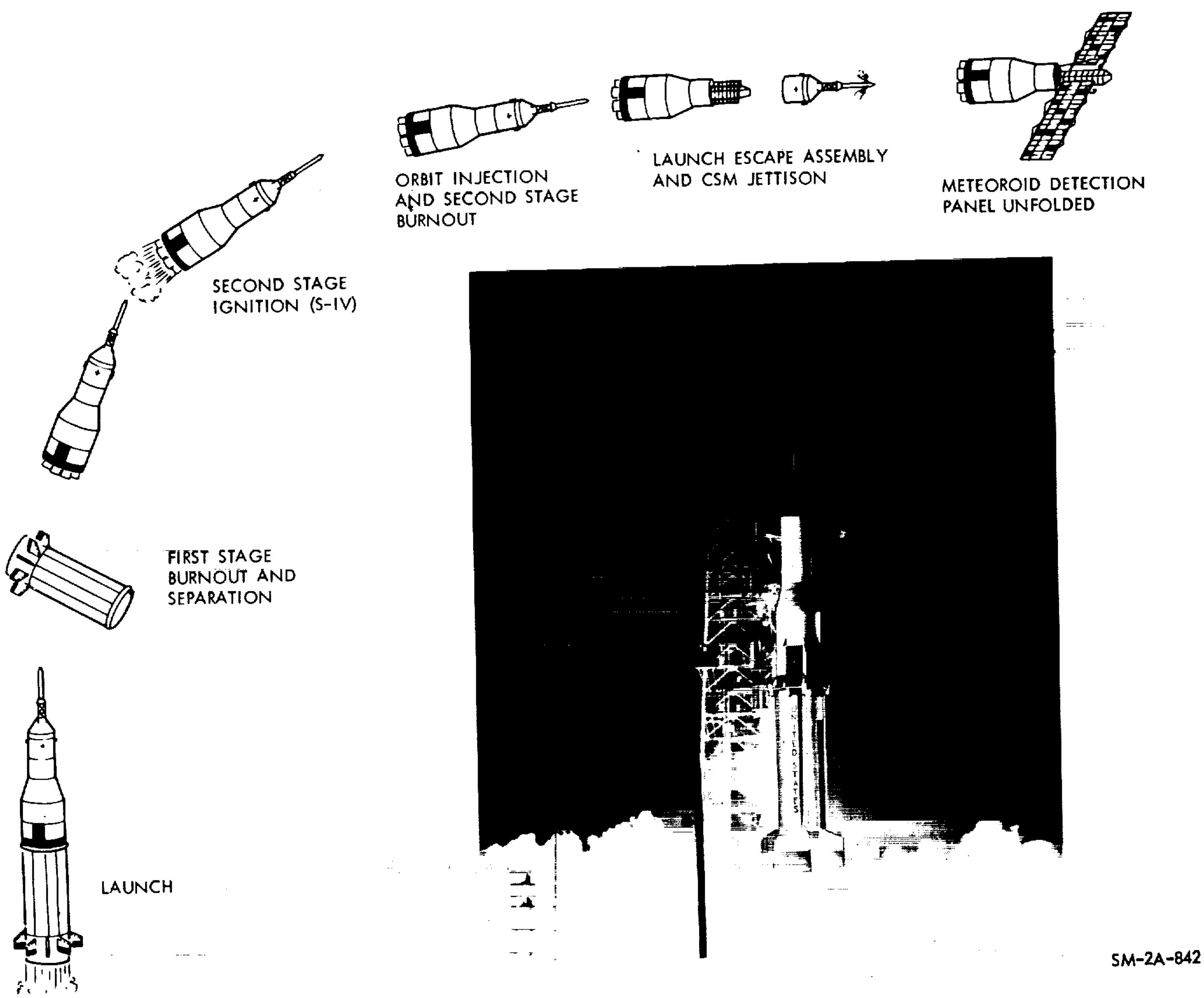


Figure 7-14. Boilerplate 26 Mission Profile

7-62. BOILERPLATE 23A.

7-63. Boilerplate 23A, another pad abort test vehicle, successfully completed its mission at the White Sands Missile Range, New Mexico, 29 June 1965. (See figure 7-15.) This was the second test of the launch escape systems ability to lift the C/M off the pad. The test simulated an emergency abort which might occur while the C/M was still on the launch pad atop a Saturn launch vehicle. An abort command was initiated causing the launch escape and pitch control motors to ignite, the C/M to separate from the S/M, and lifting the C/M from the launch pad adpater. Improvements incorporated in boilerplate 23A which were not included on the first pad abort test vehicle were: canard surfaces, boost protective cover, a jettisonable forward compartment heat shield, and reefed dual drogue parachutes. All systems worked as predicted and the C/M was lowered to the ground safely.

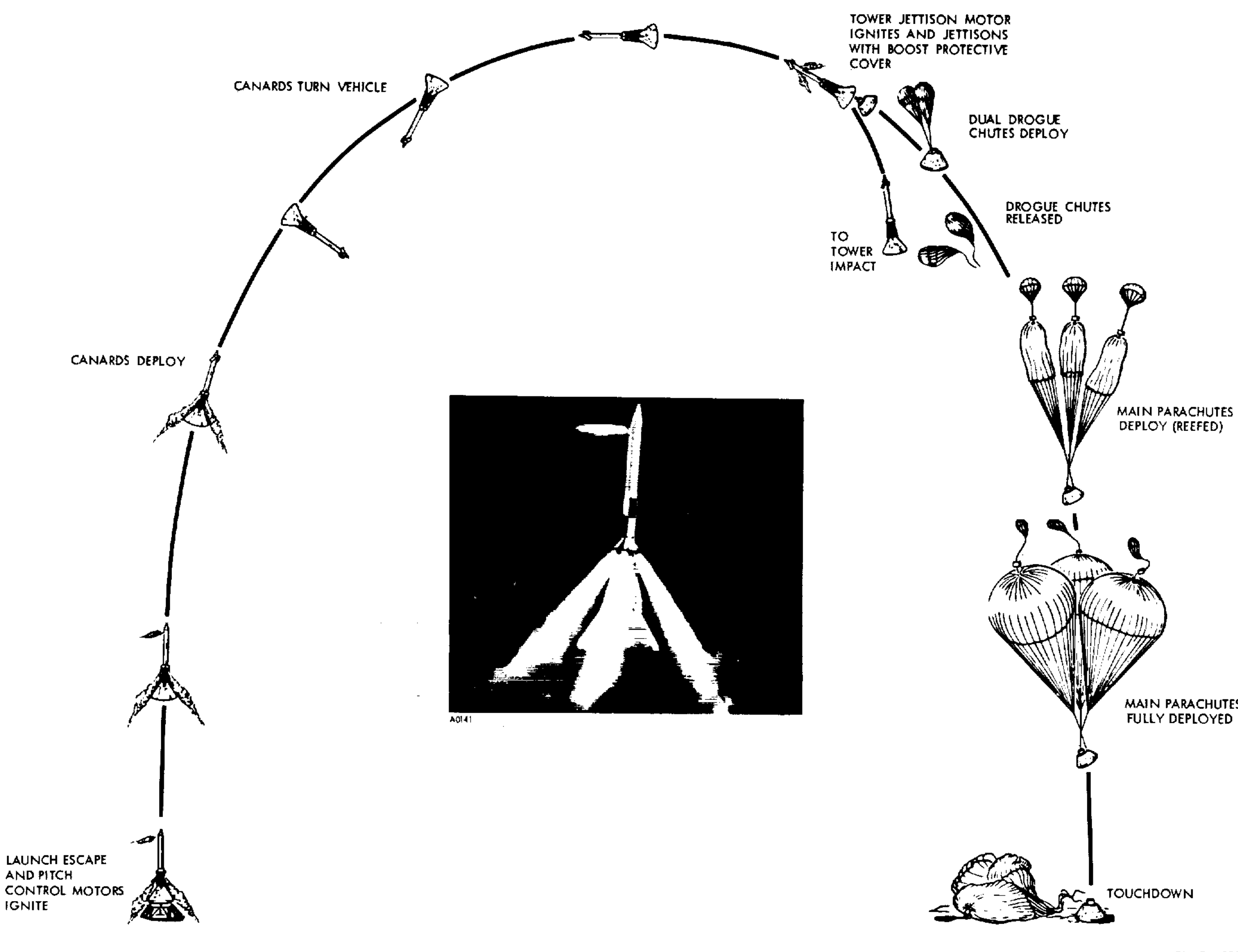


Figure 7-15. Boilerplate 23A Mission Profile

7-64. BOILERPLATE 9A.

7-65. Placing the third micrometeoroid experiment test vehicle successfully into orbit, was accomplished 30 July 1965, from Kennedy Space Center, Florida. Boilerplate 9A was used as a cover for the folded test panels, and a Saturn I with an S-IV second stage was used as the launch vehicle. (See figure 7-16.) After reaching its orbit, the CSM was jettisoned using the LES, and the large (NASA-installed) panels unfolded from the second stage (S-IV), the same as the two previous micrometeoroid test vehicles. Micrometeoroid particles are detected when they strike the panels, and the information is transmitted to ground stations by way of electronics installed in the S-IV. The test vehicle will not be recovered upon entry into the atmosphere of the earth.

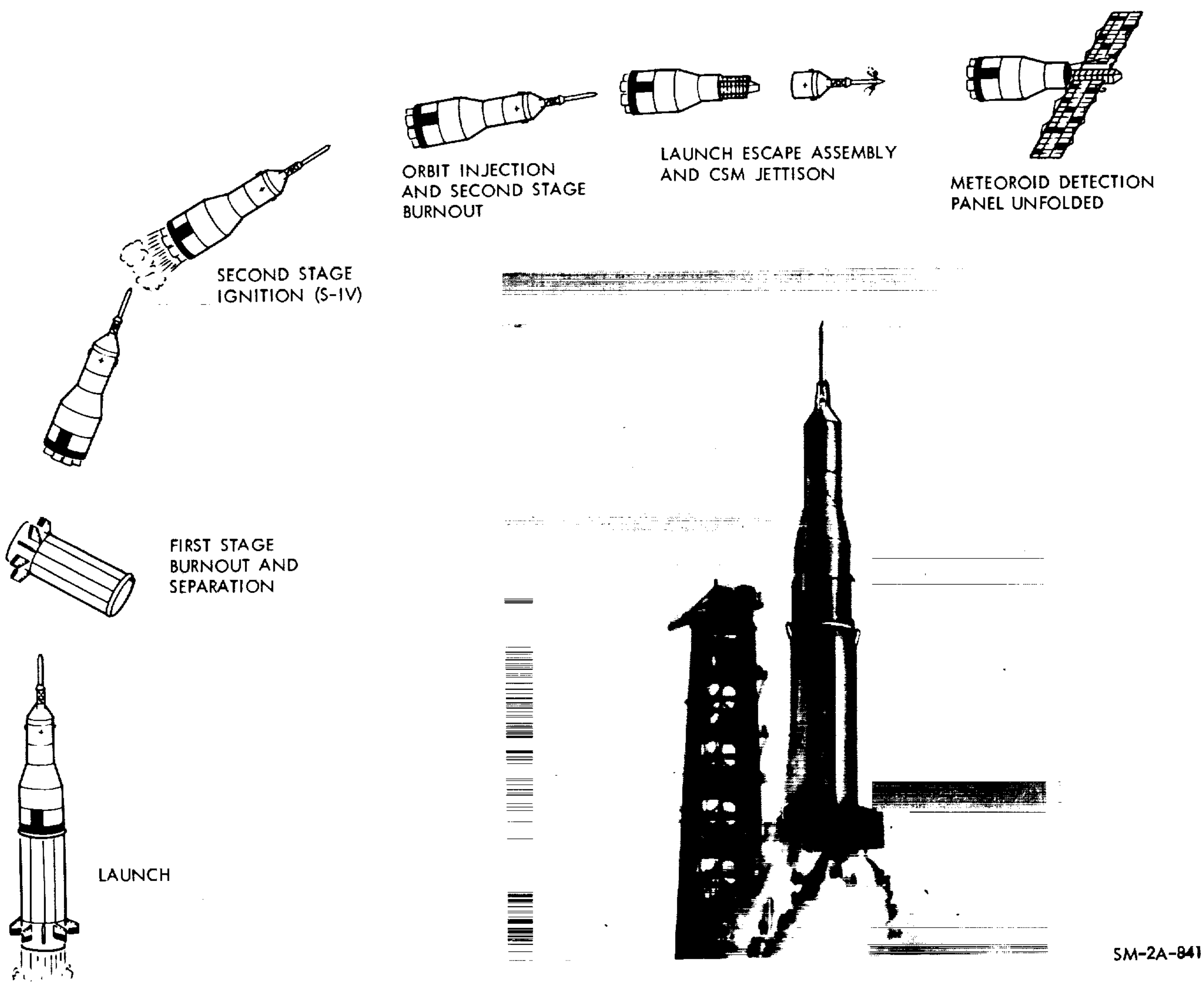


Figure 7-16. Boilerplate 9A Mission Profile

7-66. MISSIONS IN THE NEAR FUTURE.

7-67. Scheduled Apollo spacecraft missions, prior to the next December 1966 printing of this manual, are as follows:

SC 002
SC 009
SC 010
SC 011
SC 012

Relative mission and configuration data for the above spacecraft is presented in paragraph 7-23 and figure 7-3.

N

E

W

S

LUNAR DATA

DISTANCE FROM EARTH - 253,000 MILES (MAX.)

DIAMETER 2160 MILES

TEMPERATURE
SUN AT ZENITH 214°F (101°C)
NIGHT APPROX. - 250°F (-157°C)

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LUNAR LANDING MISSION

8-1. GENERAL.

8-2. The culmination of the Apollo program will be the lunar landing mission. This mission will produce the first extraterrestrial-manned exploration of the moon. This section contains a sequential presentation of the major events of the lunar landing mission. Text and illustrations within this section (figures 8-1 through 8-23) provide general information concerning the operations involved in the lunar landing mission. Text and corresponding illustrations are connected by the use of common titles.

8-3. KENNEDY SPACE CENTER.

8-4. The lunar landing mission will originate from the Kennedy Space Center (KSC). Facilities are being constructed at KSC to handle space-exploration vehicles and associated equipment. These special facilities provide capability to handle the large vehicles and components within precise parameters. Figure 8-1 shows launch pad B of complex 39 in the foreground, the remote blockhouse to the left, the 500-foot-plus vertical assembly building (VAB) in the background, and the interconnecting crawlerway.

8-5. The component assemblies of the Apollo spacecraft and the Saturn V launch vehicle will be transported to KSC for final assembly tests. The spacecraft and launch vehicle will be assembled (stacked) on the launch umbilical tower (LUT) platform in the vertical assembly building. The launch umbilical tower platform is mounted on the crawler-transporter. After assembly is completed, interface and systems tests will be made.

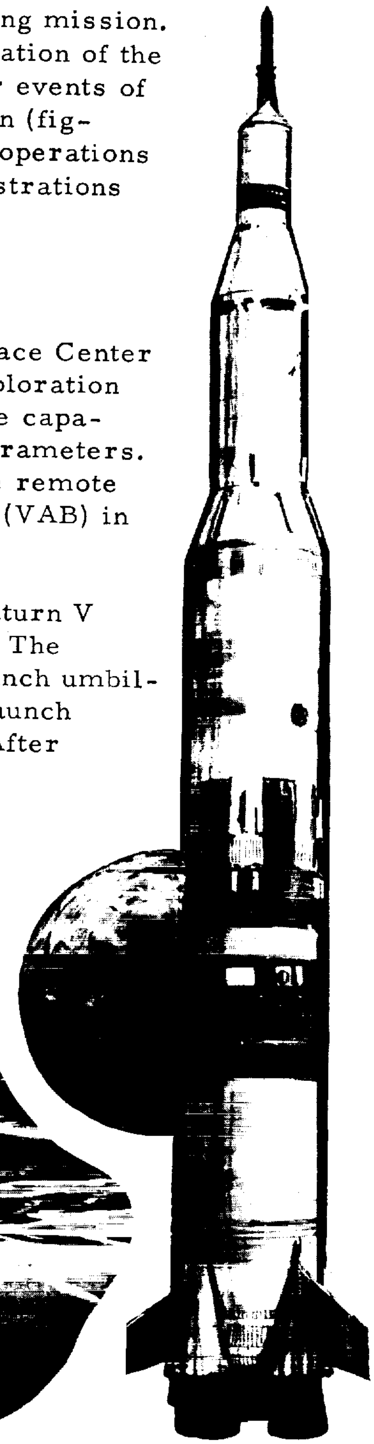

Figure 8-1. Kennedy Space Center (KSC)

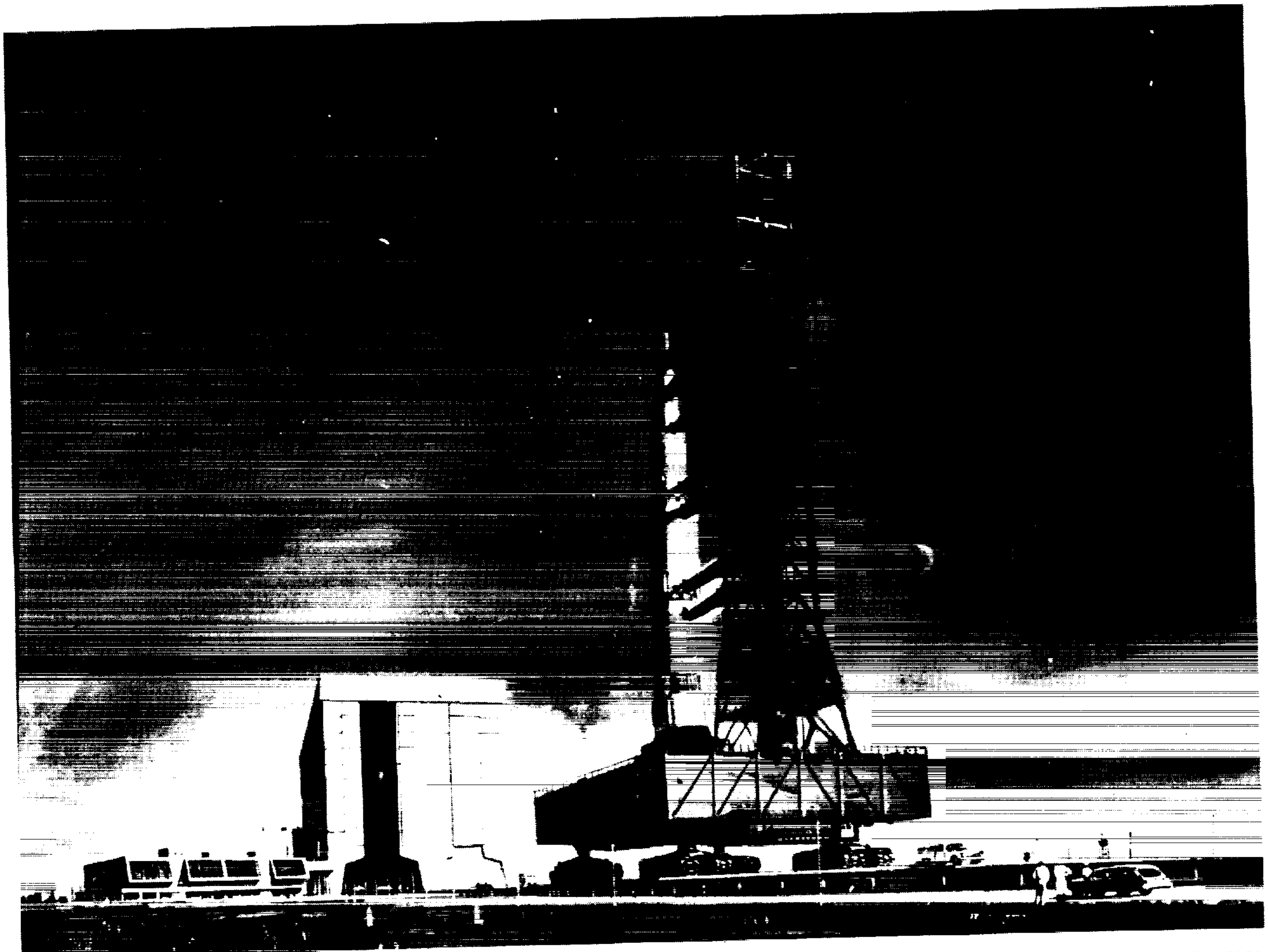

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Figure 8-2. Transportation to Launch Pad

8-6. TRANSPORTATION TO LAUNCH PAD.

8-7. After operations are concluded in the vertical assembly building, the assembled LUT, spacecraft, and launch vehicle will be transported to launch complex 39. Transportation of the launch umbilical tower, spacecraft, and launch vehicle will be provided by the crawler-transporter. The crawler-transporter will carry this load 6 miles to the launch pad on a specially constructed crawlerway. The crawlerway is a pair of parallel roadways which can support a load in excess of 17-million pounds. The crawler-transporter will proceed at a rate of approximately 1 mile per hour.

8-8. LAUNCH PAD.

8-9. Upon arrival at the launch pad, the crawler-transporter will lower the launch umbilical tower, platform, spacecraft, and launch vehicle onto steel foundations. The crawler-transporter will move an arming tower onto the pad next to the spacecraft. The arming tower provides facilities for pyrotechnic arming and fueling operations. When the arming tower is no longer needed, the crawler-transporter and the arming tower will be removed from the launch area.

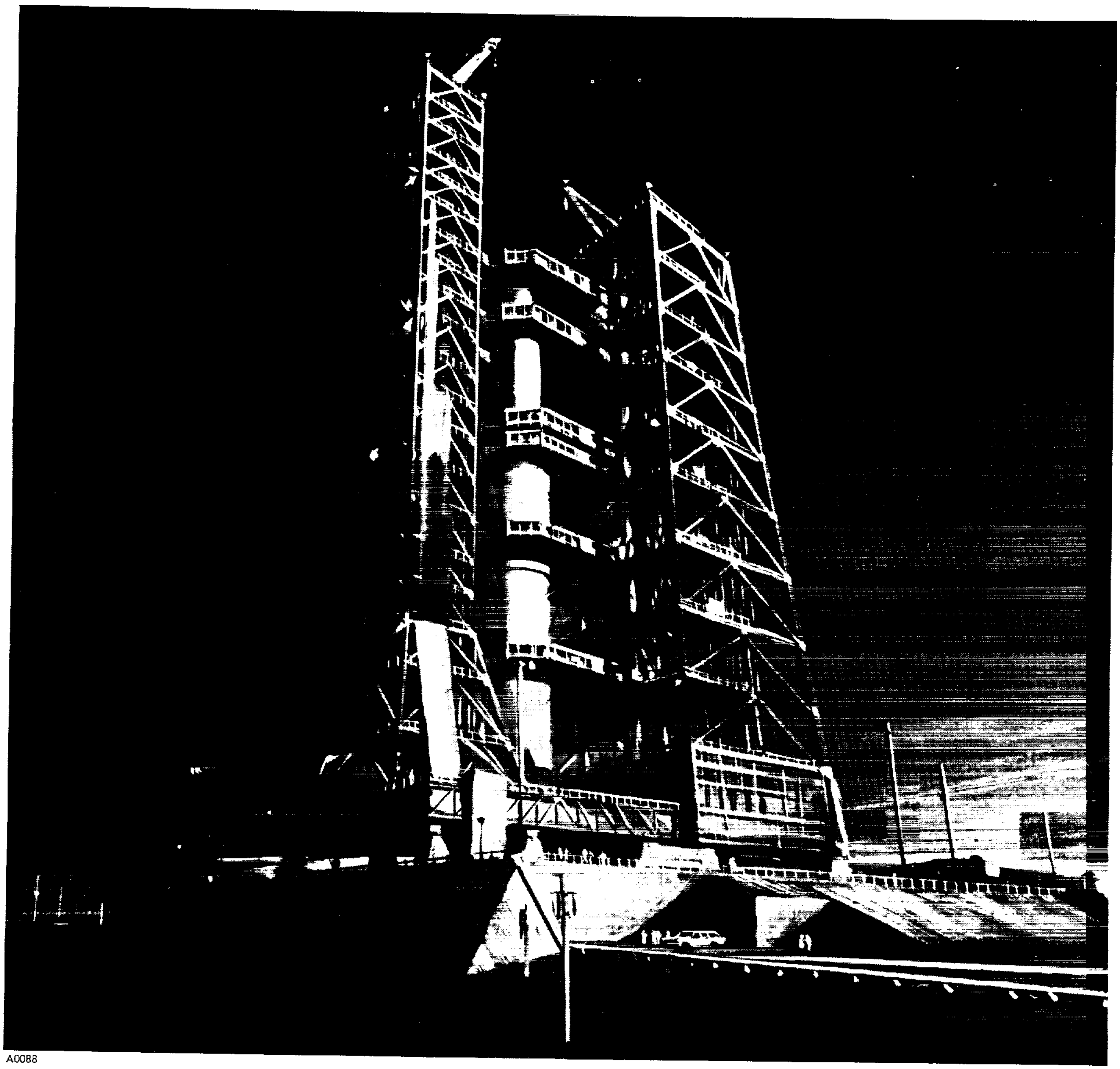

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Figure 8-3. Launch Pad

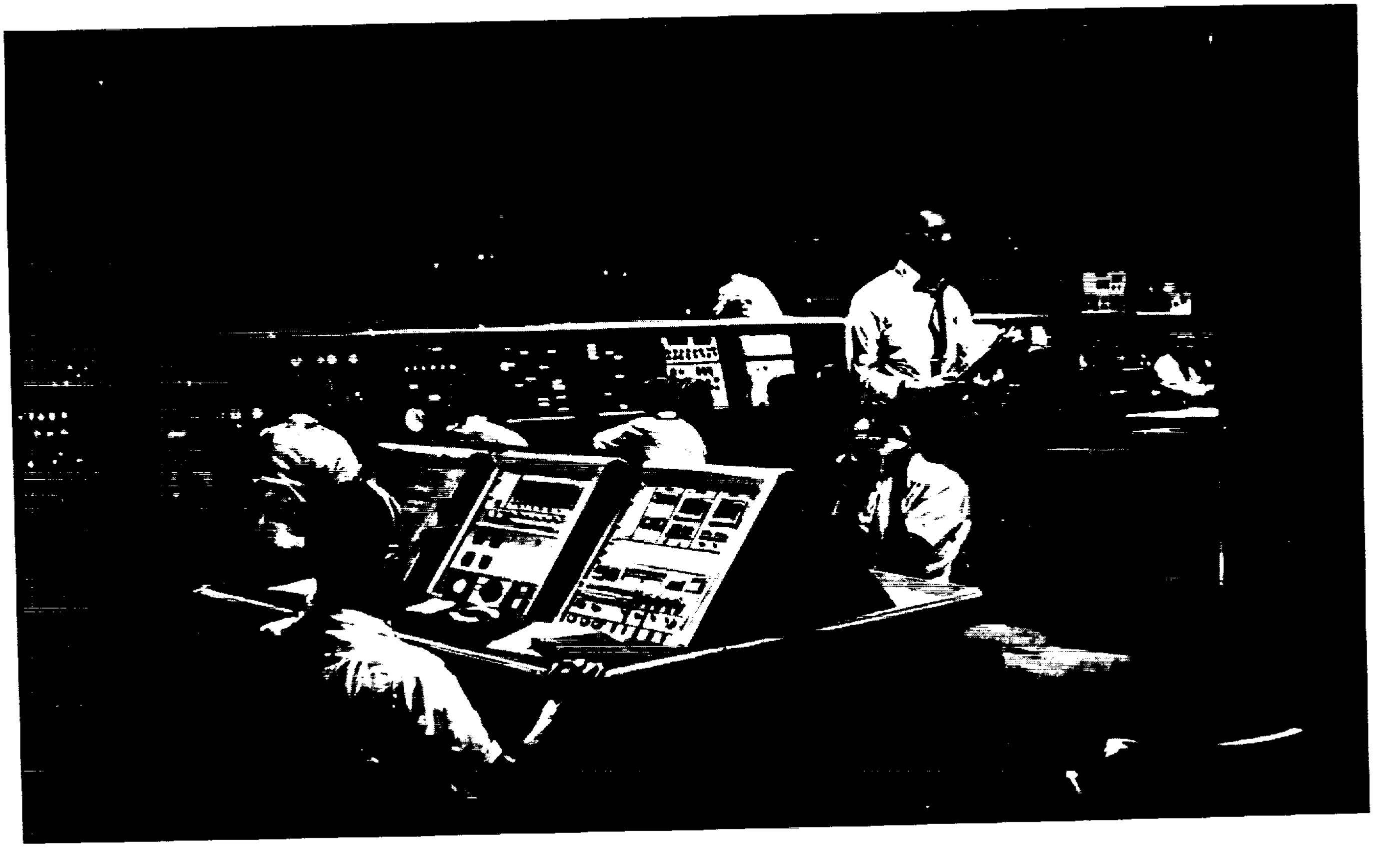

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Figure 8-4. Countdown

8-10. COUNTDOWN.

8-11. The final prelaunch countdown sequence begins upon final positioning of the launch umbilical tower, spacecraft, and launch vehicle on the launch pad. Ordnance components required on the spacecraft are armed, utilizing the arming tower. Appropriate protective devices are installed at the time of ordnance installation to prevent inadvertent operational arming and firing of the pyrotechnics, and to provide maximum safety for the spacecraft checkout crew and launch area ground personnel.

8-12. The prelaunch countdown follows a programed sequence which is directed by, and under the control of, the launch control director. This sequence establishes the order of required operational checkout of the spacecraft systems and of the servicing and loading of consumable gases, fuels, and supplies.

8-13. The countdown sequence consists essentially of the activation, or simulated activation, and verification checks of the spacecraft operational systems as follows:

- Removal of ground support equipment (GSE)
- Leak checks
- Battery activation
- Final arming of ordnance devices
- Removal or ordnance shorting devices
- Loading of fuels: helium, liquid hydrogen, and liquid oxygen
- Fuel cell activation
- Entry of mission flight crew into command module
- Closing of command module crew hatch
- Installation of boost protective cover hatch access cover
- Command module crew cabin leak check
- Purging the command module cabin with 100-percent oxygen
- Final confidence checks of the spacecraft systems by the crew
- Final arming of the launch escape system
- Ground-to-spacecraft umbilical disconnect.

8-14. Upon completion of final checks, the ground-to-spacecraft umbilical cables are disconnected and the launch tower support arms are retracted. Final decision and approval to launch is verified by the launch control center and the spacecraft crew.

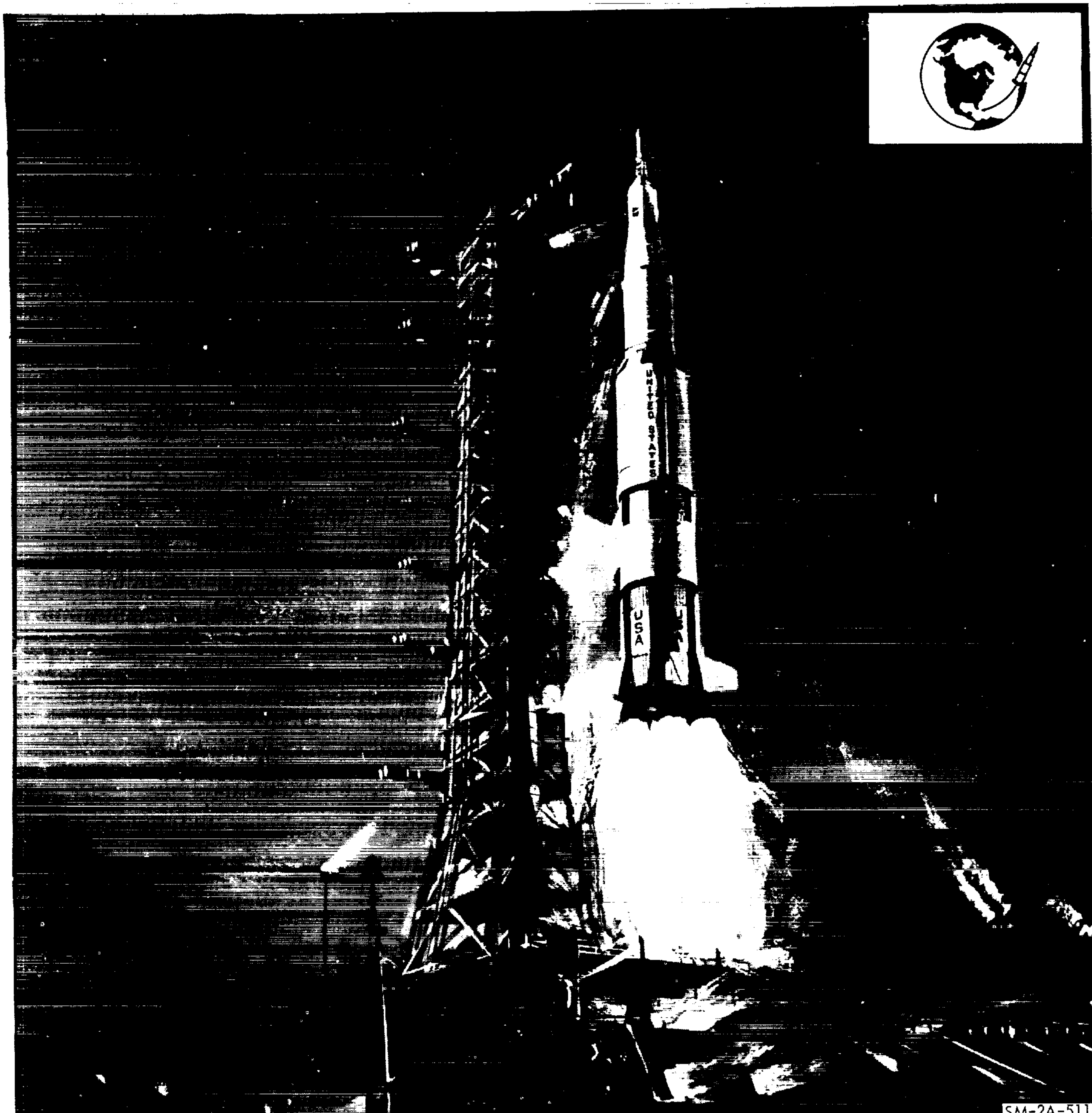

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Figure 8-5. Lift-Off

8-15. LIFT-OFF.

8-16. Upon launch, the Saturn V first-stage (S-IC) engines are ignited by the launch control center. The center engine ignites first, followed by the ignition of the four outer engines. The launch pad hold-down devices release after initial operational thrust is sufficient. The launch control center and the spacecraft crew will continuously monitor the initial operational ascent attitude parameters.

8-17. FIRST-STAGE SEPARATION.

8-18. The first-stage guidance system initiates roll of the spacecraft to the required launch azimuth. The first-stage pitch programer initiates the required pitchover of the spacecraft. Voice communication between the crew and manned space-flight network (MSFN) is maintained through the critical maximum dynamic flight conditions and throughout the ascent phase. The cutoff of the first-stage engines is followed by ignition of the second-stage ullage rockets. The first-stage retrorocket then separates the first stage from the second stage.

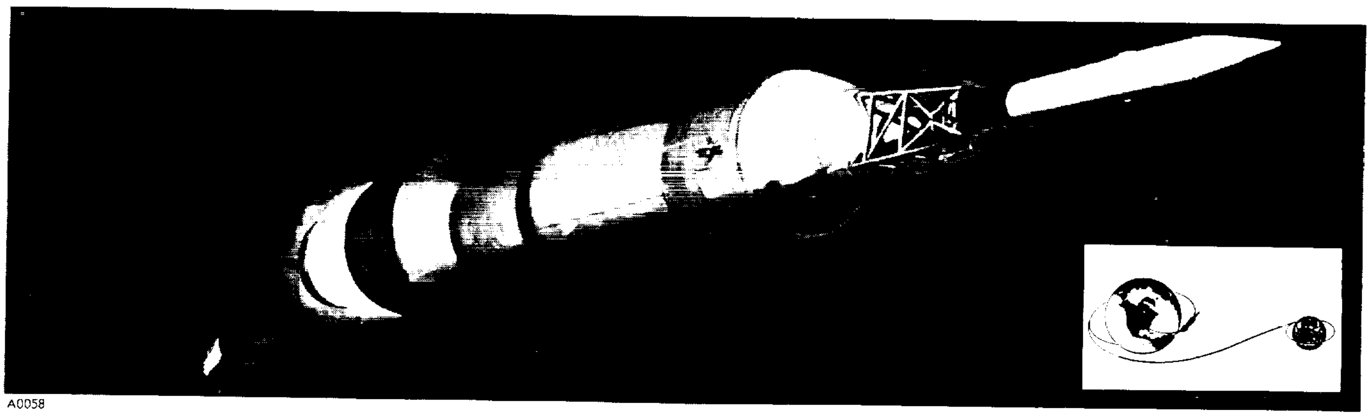

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Figure 8-6. First-Stage Separation

8-19. SECOND-STAGE EVENTS.

8-20. Second-stage (S-II) engine ignition occurs nominally two seconds after cutoff of the first-stage engines at approximately 200,000 feet. The flight trajectory of the spacecraft is controlled by the S-IVB inertial guidance system. The launch escape system is operationally jettisoned at approximately 300,000 feet altitude. The second-stage engines are cut off at an altitude of approximately 600,000 feet. The sequential ignition of the third-stage (S-IVB) ullage rockets, second-stage retrorockets, and third-stage engines effects separation of the second stage. The third-stage engines provide the thrust required to place the spacecraft in earth orbit. The third-stage guidance control system cuts off the third-stage engines after the programed orbit conditions have been attained.

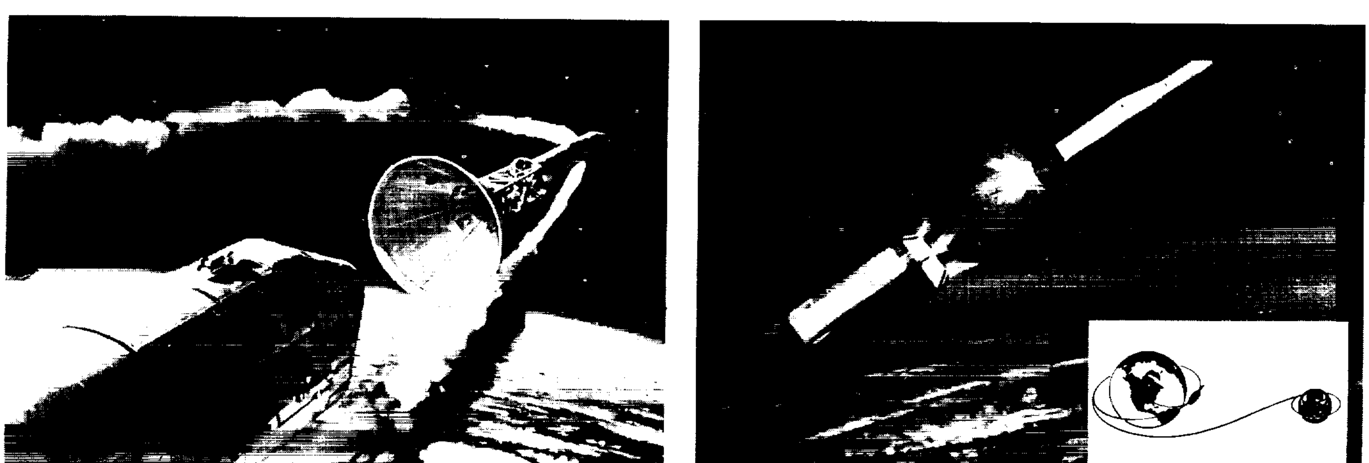

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Figure 8-7. Second-Stage Events

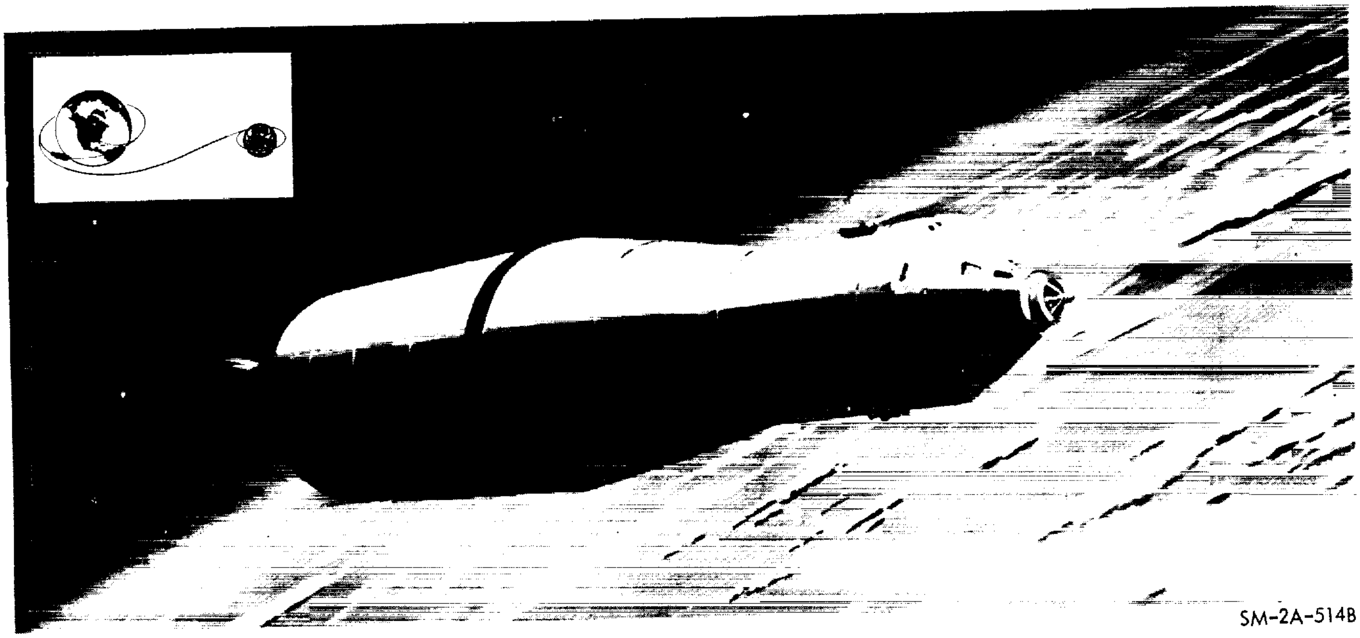

Figure 8-8. Earth Orbit

8-21. EARTH ORBIT.

8-22. The spacecraft and third stage are to orbit the earth, no more than three times, at an approximate altitude of 100 nautical miles. During this period, the orbital parameters are determined by manned space-flight network; then verified by landmark navigational sightings and the spacecraft crew. This determines the required velocity increment and trajectory for translunar injection.

8-23. The crew will perform a biomedical and safety equipment check. Sequence checks will be made of the environmental control system, communications and instrumentation system, service propulsion system, service module reaction control system, electrical power system, guidance and navigation system, stabilization and control system, and crew equipment system.

8-24. Translunar injection parameters are determined onboard the spacecraft by sequential landmark navigational sightings (using the scanning telescope) and by the Apollo guidance computer. Trajectory and star-tracking data computations are made by the Apollo guidance computer. The inertial measurement unit is fine-aligned for the translunar injection monitor, using the Apollo guidance computer. The center-of-gravity offset angles are set into the service propulsion system gimbal position display. The ΔV program, time, and direction vector are set into the Apollo guidance computer. The stabilization and control system is prepared for the ΔV maneuver, including minimum deadband hold control and monitor mode. Finally, the third-stage reaction control system is prepared for the ΔV translunar injection.

8-25. Verification of "go" conditions for translunar injection will be confirmed by the spacecraft crew and the MSFN. The third-stage countdown and ignition sequence is performed with the spacecraft in the required translunar injection attitude.

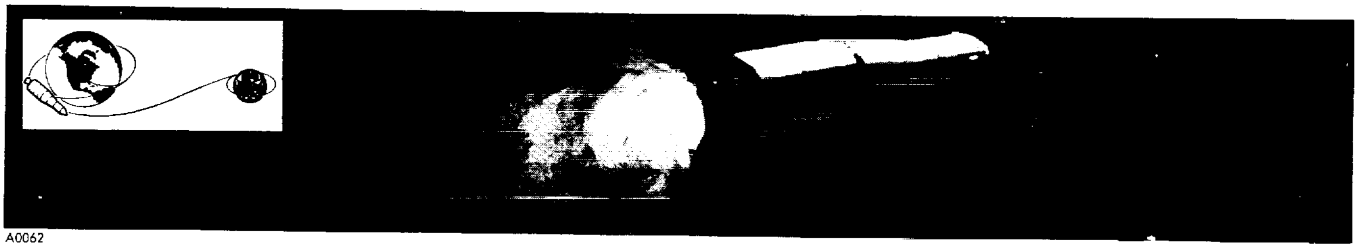

Figure 8-9. Translunar Injection

8-26. TRANSLUNAR INJECTION.

8-27. The translunar injection phase begins with the third-stage ullage rockett ignition. The third-stage propulsion system is operated to provide sufficient thrust to place the spacecraft in a translunar "free-return" trajectory in accordance with the ΔV magnitude, time duration, and thrust vector previously established and operationally programed onboard the spacecraft by the crew, and confirmed by the MSFN.

8-28. The third-stage instrument unit provides operational guidance control for the translunar injection with the Apollo guidance and navigation system capable of backup control, if necessary. The third-stage engines operate for the predetermined time, nominally 5 minutes. The crew monitors the emergency detection system and spacecraft attitude control displays. The spacecraft guidance and navigation system monitors the programed injection maneuver.

8-29. INITIAL TRANSLUNAR COAST.

8-30. Following translunar injection, the MSFN will determine the spacecraft trajectory and verify it with an onboard determination performed by the crew. The operational controls are then set for an initial coast phase. An onboard systems check is then made of all crew equipment, electrical power system, environmental control system, service module reaction control system, and the service propulsion system. The status of these systems is communicated to MSFN.

8-31. The spacecraft body-mounted attitude gyros are aligned, using the third-stage stable platform as a reference, and the flight director attitude indicator is set preparatory to initiating transposition of the lunar excursion module. Confirmation of conditions for initiating transposition of the lunar excursion module is made with MSFN.

Figure 8-10. Initial Translunar Coast

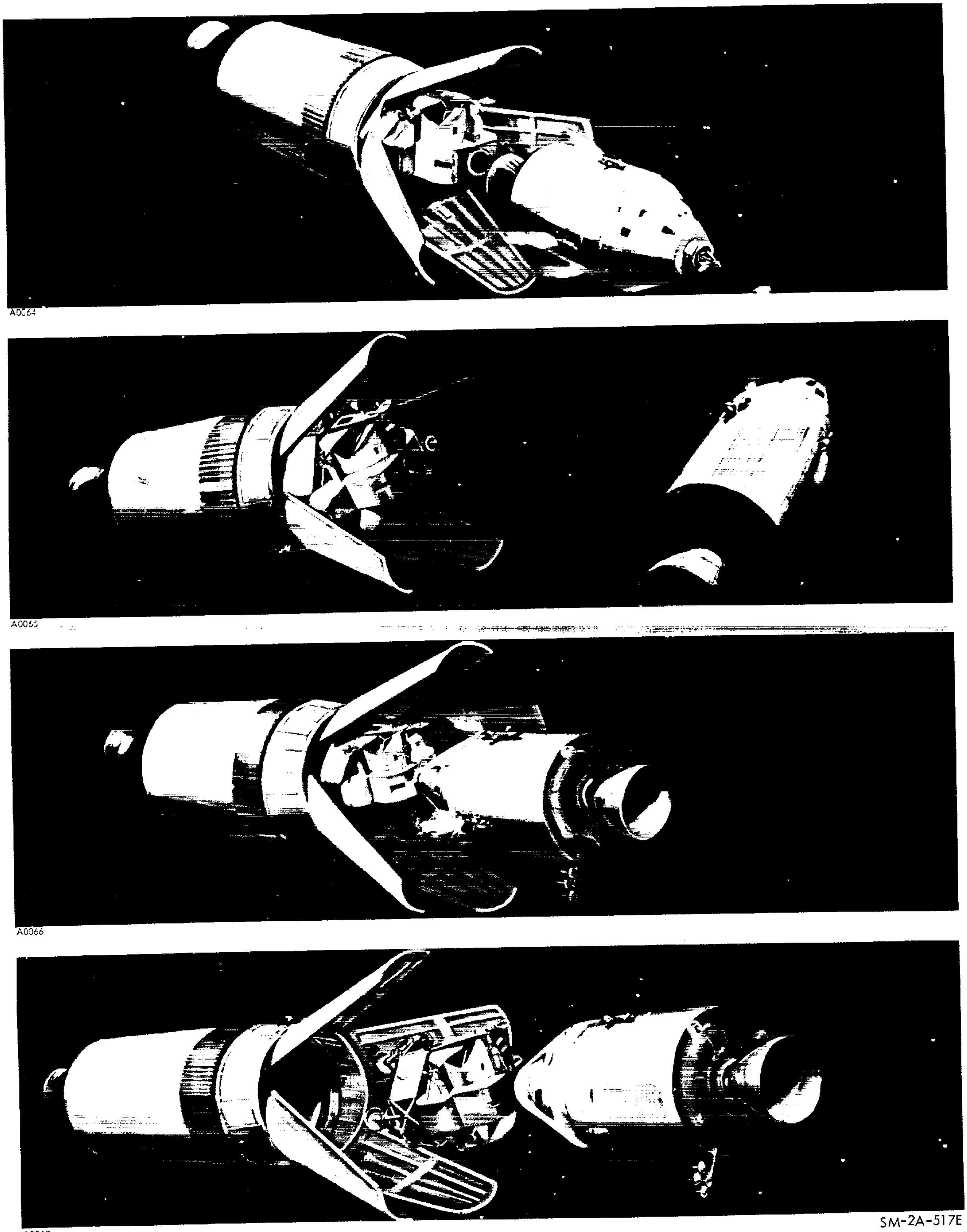

Figure 8-11. Lunar Excursion Module Transposition

8-32. LUNAR EXCURSION MODULE TRANSPOSITION.

8-33. Transposition of the lunar excursion module consists essentially of separating and translating the command/service module from the spacecraft-LEM-adapter (SLA), third stage, pitching the command/service module 180 degrees, and translating the command/ service module back to the lunar excursion module to join the lunar excursion module to the command module. The S-IVB guidance system stabilizes the SLA, third stage, during the transposition operations. Upon completion of docking, the third stage is jettisoned. The entire maneuver will normally be completed within one hour after translunar injection. Necessary precautions will be observed by the crew during the time the spacecraft passes through the Van Allen belts.

8-34. The spacecraft will be oriented within communication constraints to provide the most desirable background lighting conditions for the transposition of the lunar excursion module. The third stage is stabilized in an attitude-hold mode. The adapter is pyrotechnically separated from the command/service module, which is then translated approximately 50 feet ahead of the lunar excursion module and third stage, using the service module reaction control system engines. The command/service module is then rotated 180 degrees in pitch, using the service module reaction control engines. The docking attitude of the command/service module for the SLA, third stage will be established and maintained, using the service module reaction control system engines.

8-35. The command/service module will be translated toward the SLA, third stage, with minimum closing velocity, so that the command/service module probe engages within the drogue mechanism on the lunar excursion module. A mechanical latching assembly secures the lunar excursion module to the command/service module. The command/ service module, with the lunar excursion module attached, then separates and translates away from the third stage.

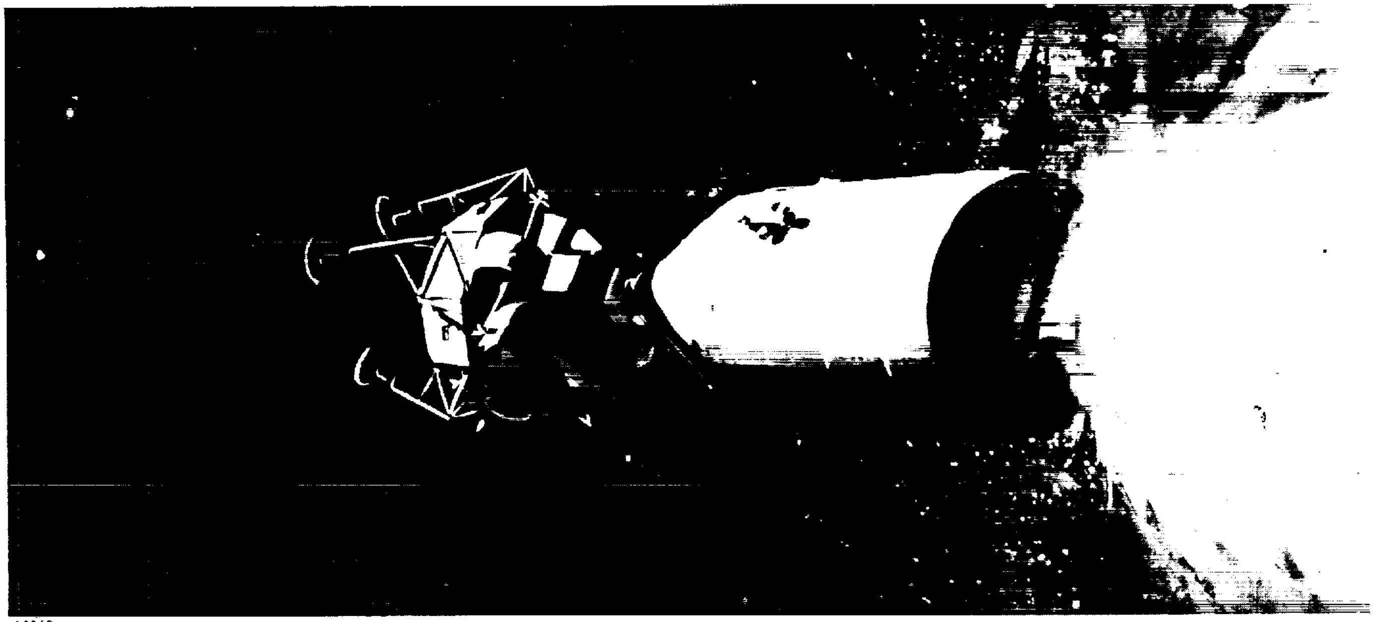

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Figure 8-12. Final Translunar Coast

8-36. FINAL TRANSLUNAR COAST.

8-37. The final translunar coast phase begins with the ignition of the service module reaction control system to separate the spacecraft from the third stage, and ends just prior to lunar orbit insertion. The primary operations occurring during this phase consist of system checkout of the spacecraft systems, trajectory verifications, and preparation for lunar orbit insertion. Midcourse ΔV corrections, navigational sightings, and inertial measurement-unit alignments are to be made if needed. A ΔV budget sufficient to provide a total ΔV correction of 160 feet per second is planned.

8-38. The spacecraft guidance and navigation system computes the trajectory of the spacecraft (in conjunction with navigational sightings). The delta increment required is determined by MSFN, and confirmation of the trajectory and velocity increment values is made with the Apollo guidance computer. Midcourse incremental velocity corrections will be made if required.

8-39. The attitude of the spacecraft will be constrained at times because of operational temperature control restrictions. At least one astronaut will be in his space suit at all times. A crew work-rest cycle will be established and followed during this phase. The capability to initiate an abort at any time during this phase will be provided.

8-40. In preparation for lunar orbit insertion, the spacecraft attitude, lunar orbit insertion velocity increment, and the time to initiate the service propulsion system thrust required to achieve the desired orbit around the moon are determined by trajectory data from MSFN and from the guidance and navigation system.

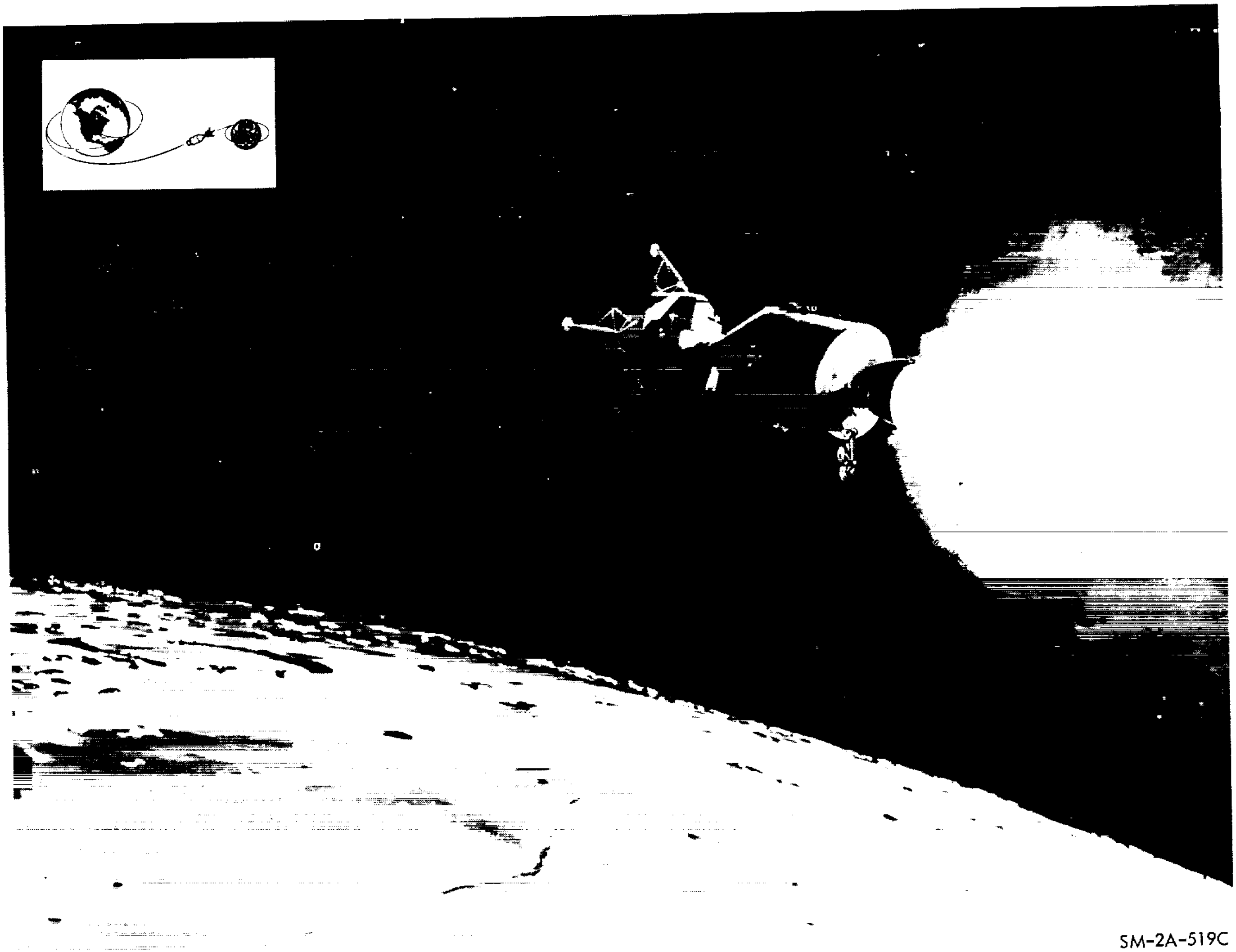

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Figure 8-13. Lunar Orbit Insertion

8-41. LUNAR ORBIT INSERTION.

8-42. This phase begins with the spacecraft properly oriented for lunar orbit insertion and ends with the cutoff of the service propulsion system as the spacecraft is inserted into a lunar orbit.

8-43. Navigation sightings will be made using the Apollo guidance computer, inertial measurement unit, and scanning telescope. The MSFN determines the lunar orbit insertion trajectory and star catalog data, ΔV correction parameters, and the lunar orbit insertion parameters. These determinations are confirmed using the Apollo guidance computer. The reaction control system ignition provides a translation impulse and roll-control operation. The guidance and navigation system initiates and controls the programed insertion maneuvers and service propulsion system.

8-44. The retrograde impulse required to establish the lunar orbit occurs near the minimum altitude of the lunar approach trajectory. The point of this altitude is almost directly behind the moon with respect to the earth. The total velocity increment required to achieve the desired orbit around the moon, including any necessary plane changes, is applied during this phase.

8-45. The initial lunar orbit coast phase begins with cutoff of the service propulsion engine as the spacecraft is inserted into lunar orbit and ends with activation of the lunar excursion module reaction control system to effect separation from the spacecraft.

8-46. Following lunar orbit insertion, the crew will transmit trajectory data and information to MSFN. The orbit ephemeris about the moon will be determined as accurately as possible, using the spacecraft guidance system and MSFN. A confirming checkout of the lunar excursion module guidance system is also made prior to separation from the spacecraft.

8-47. The spacecraft systems will be capable of operation at their nominal design performance level for a mission of approximately 11 days. A single crewmember can control the spacecraft in lunar orbit for several days. Communication capability will be provided between the spacecraft, manned space-flight network, and the lunar excursion module when separated and within line-of-sight. The spacecraft and lunar excursion module separation and docking operations will not be restricted by natural illumination conditions.

8-48. Observations and calculations will be made of the preselected landing site from the spacecraft, to determine if the area location is satisfactory or if an alternate landing area should be selected. Detailed surveillance of the landing area is to be made from the lunar excursion module prior to landing.

8-49. Lunar orbit trajectory verification requires fine-alignment of the inertial measurement unit, and a related series of navigational sightings will be made of known lunar surface areas and reference stars, using the scanning telescope, sextant, and Apollo guidance computer. Parameters for lunar orbit and transearth injection will be determined by MSFN and confirmed by the Apollo guidance computer; but the LEM descent trajectory will be calculated using the LEM guidance, navigation, and control system.

8-50. Upon final confirmation of these parameters, the commander and the systems engineer will transfer from the spacecraft to the lunar excursion module. The lunar excursion module electrical power system, environmental control system, communication system, guidance, navigation, and control system, reaction control system, and ascent and descent engine systems will be checked out and the landing gear extended. A check will be made of emergency procedures and corresponding spacecraft systems. The operational capability of the air lock will be verified. Initial operational information will be synchronized between the spacecraft and lunar excursion module.

8-51. The spacecraft will be aligned and held in the required attitude for separation. The lunar excursion module guidance computer will be programed for the transfer trajectory when the spacecraft orbit is determined accurately. The orbit will not be disturbed, unless an emergency requirement prevails, or until the docking phase is complete. Emergency or additional data may require that the spacecraft lunar orbit be updated as necessary by the remaining crewmember.

8-52. Actual separation of the module from the spacecraft is effected by a propulsion thrust from the lunar excursion module reaction control system. After a specified time, an equivalent impulse is applied in the opposite direction so that the relative velocity between the lunar excursion module and the spacecraft will be zero during the final checkout of the lunar excursion module. Final checkout is accomplished with the lunar excursion module in free flight, but relatively close to the spacecraft in case immediate docking is required.

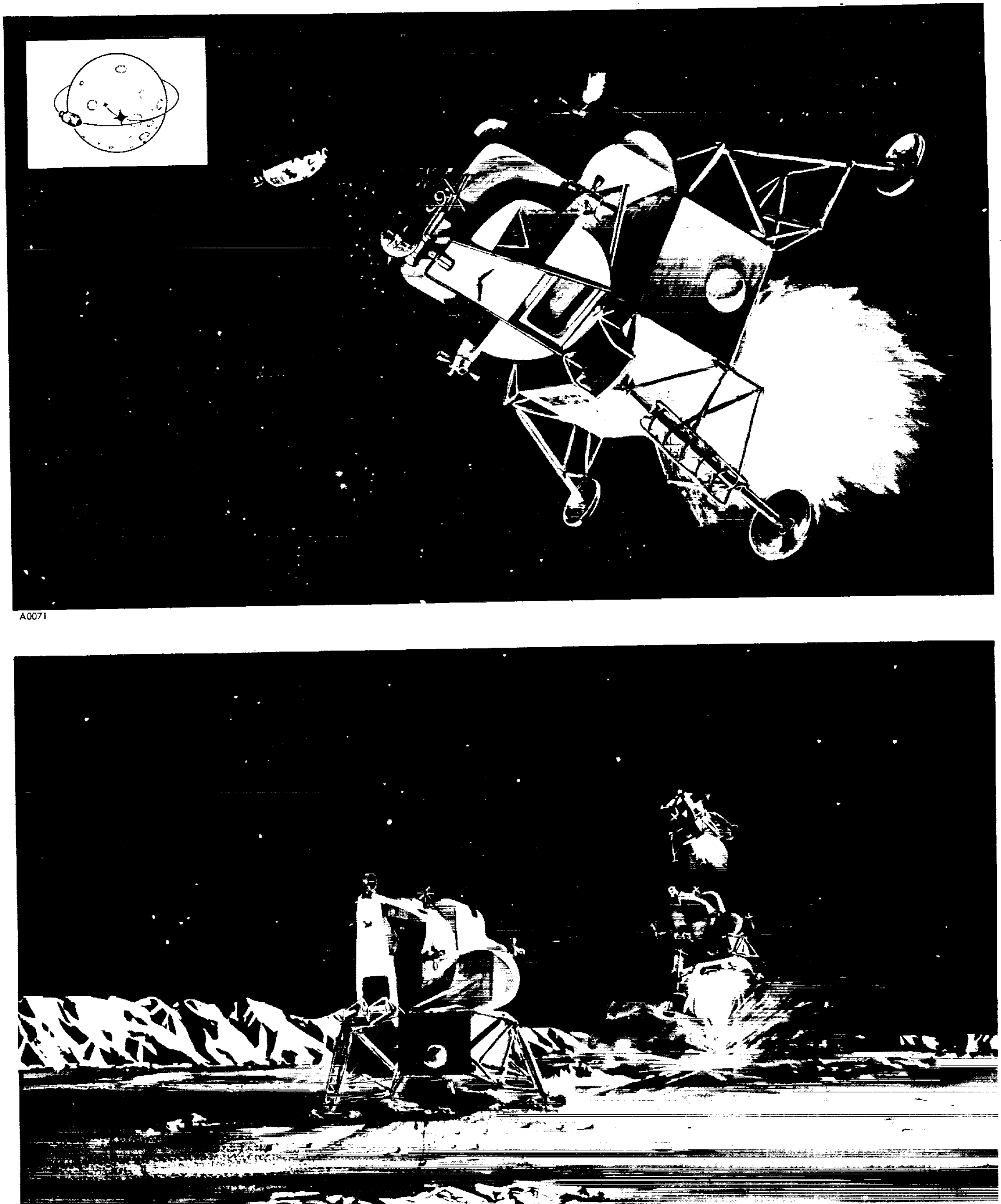

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Figure 8-14. Lunar Landing

8-53. LUNAR LANDING.

8-54. The lunar excursion module lunar landing operations phase begins with initial lunar excursion module separation from the spacecraft and ends with touchdown of the lunar excursion module on the surface of the moon.

8-55. The essential lunar excursion module operations which occur are attitude control, incremental velocity control, and time to fire, computed by the lunar excursion module guidance, navigation, and control system and verified by the spacecraft guidance and navigation system. Lunar excursion module insertion into a descent orbit is accomplished by reaction control system ullage acceleration and ignition of the descent engine with the thrust level and burn-time automatically controlled by the lunar excursion module guidance, navigation, and control system. The lunar excursion module separates from the spacecraft during this maneuver, and communication with manned space-flight network at time of insertion cannot be accomplished, since it occurs on the far side of the moon.

8-56. Descent trajectory determination will be made using data from the LEM rendezvous radar tracking the spacecraft. The lunar excursion module will coast in descent transfer orbit following descent engine cutoff, and close observation of the proposed lunar landing site will be made for final approval. The descent engine will be re-ignited prior to reaching the low point in the orbit and sustained thrust initiated for the descent maneuver. The thrust and attitude are controlled by the guidance, navigation and control system by comparison of the actual and planned landing tracks.

8-57. The translational and radial velocities will be reduced to small values and the descent engine cut off at a specified altitude above the lunar surface. The commander will control the descent within present limits. Terminal descent and touchdown will be made by manual control and use of the landing radar. Confirmation of initial lunar touchdown will be made by the lunar excursion module crew to the spacecraft and to MSFN.

8-58. The crewman in the spacecraft will maintain visual observation of the lunar landing operation as long as possible. All three crewmen will be in their space suits.

8-59. LUNAR SURFACE OPERATIONS.

8-60. The lunar surface operations phase begins with lunar touchdown and ends with launching of the lunar excursion module from the moon.

8-61. Initial tasks to be performed by the two astronauts following touchdown include review and determination of the lunar ascent sequence and of the parameters required. A complete check of the lunar excursion module systems and structure will be made. Necessary maintenance will be determined and performed to assure the operational ascent capability of the lunar excursion module. The systems will be put into a lunar-stay mode and a systems monitoring procedure established. The lunar excursion module will be effectively secured as necessary, and the landing and launch stage disconnect mechanisms activated.

8-62. The lunar landing must be made on the earth-side of the moon to permit and establish communication with MSFN and the lunar orbiting spacecraft from the surface of the moon. Voice and signal communication will be verified prior to beginning egress and lunar exploration activity. A post-touchdown status report will be made to the spacecraft before line-of-sight communication is lost as the spacecraft orbits below the lunar horizon. The position and attitude of the lunar excursion module on the moon will be established and reported.

8-63. The lunar excursion module is capable of operating normally on the lunar surface during any phase of the lunar day-night cycle. The lunar excursion module, designed to be left unoccupied with the cabin unpressurized on the lunar surface, will be capable of performing its operations independently of earth-based information or control.

8-64. Although the nominal lunar stay-time may be from 4 to 35 hours, depending on the planned scientific exploration program, the capability to launch at any time in an emergency situation shall be established. Portable life support systems will provide the capability for 24 man-hours of separation from the lunar excursion module. Maximum continuous separation will be 4 hours (3 hours of normal operation plus 1 hour for contingencies).

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Figure 8-15. Lunar Surface Operations

8-65. The two astronauts will descend to the lunar surface to perform scientific exploration and observation and the lunar excursion module will be left unattended. The scientific exploration activity may include gathering selected samples from the lunar surface and atmosphere, measurement of lunar surface and atmospheric phenomena, and the securing of scientific instruments on the lunar surface for signal transmission and telescopic observation from earth. Video transmission from the lunar surface may be accomplished by means of portable television equipment. Provision will be made for return of approximately 80 pounds of samples from the lunar surface.

8-66. Following completion of the lunar surface exploration activity, preparation for ascent will begin. The two astronauts will return to the LEM and secure themselves in the LEM cabin. Launch and rendezvous plans will be confirmed with the crewman in the spacecraft and with MSFN. The spacecraft tracking and rendezvous data determination sequence will be initiated. The lunar excursion module operational systems required for lunar ascent, the ascent and descent stage separation, and the inertial measurement unit alignment will be checked out.

8-67. SPACECRAFT SOLO LUNAR ORBIT OPERATION.

8-68. During separation of the lunar excursion module from the spacecraft for lunar operations, the crewmember in the spacecraft will perform a series of backup operations in support of the lunar activity.

8-69. The spacecraft crewmember will initially monitor the separation sequence and initiate optical tracking of the LEM. A communication link between the spacecraft, lunar excursion module, and MSFN will be established. The spacecraft will monitor the LEM orbit injection sequence and maintain cognizance of essential operational parameters.

8-70. In addition, periodic operational checks will be made of the spacecraft systems, the inertial measurement unit alignment procedure performed, and the lunar orbit parameters periodically updated and confirmed with MSFN.

8-71. The lunar landing sequence may be monitored by optical tracking and essential operational data will be transmitted to MSFN. The location of the lunar landing site will be determined. Visual observations will be made of the lunar surface operations and periodic line-of-site communication with the crew maintained as required.

8-72. Spacecraft confirmation of the LEM ascent and rendezvous parameters will be established. Radar tracking of the ascent trajectory will be established to permit determination of the operational maneuvers to effect rendezvous. The spacecraft guidance and navigation system determines these essential parameters.

8-73. For rendezvous and docking, the spacecraft normally will be stabilized in a passive mode. However, the spacecraft will have the capability of controlling the terminal attitude and translation required for rendezvous and docking. The spacecraft solo operation ends after completion of docking, when the crew transfers from the LEM to the spacecraft.

8-74. LUNAR ASCENT.

8-75. Confirmation of "go" conditions for lunar launch is made with the spacecraft and MSFN. The ascent engine will be ignited and launch from the lunar surface accomplished.

8-76. The lunar excursion module ascent trajectory places it in a position approximately 50,000 feet above the lunar surface, at a velocity such that the resultant orbit about the moon has a clear minimum altitude of approximately 50,000 feet and a nominal intercept with the orbiting CSM. The launch trajectory of the lunar excursion module is controlled through its guidance, navigation, and control system. The lunar excursion module rendezvous radar tracks the spacecraft during the ascent to provide inputs to the guidance, navigation, and control system. The lunar excursion module reaction control system executes the required roll, attitude, and pitch maneuvers to place it in the required orbit.

8-77. Following the cutoff of the ascent engine, the radar will continue to track the spacecraft. The spacecraft guidance and navigation system will compute the orbit of the lunar excursion module and the crew will determine if any ΔV corrections are required to effect rendezvous. The final coast trajectory parameters, range, rate, and attitude angles will be determined and rendezvous operations will be initiated.

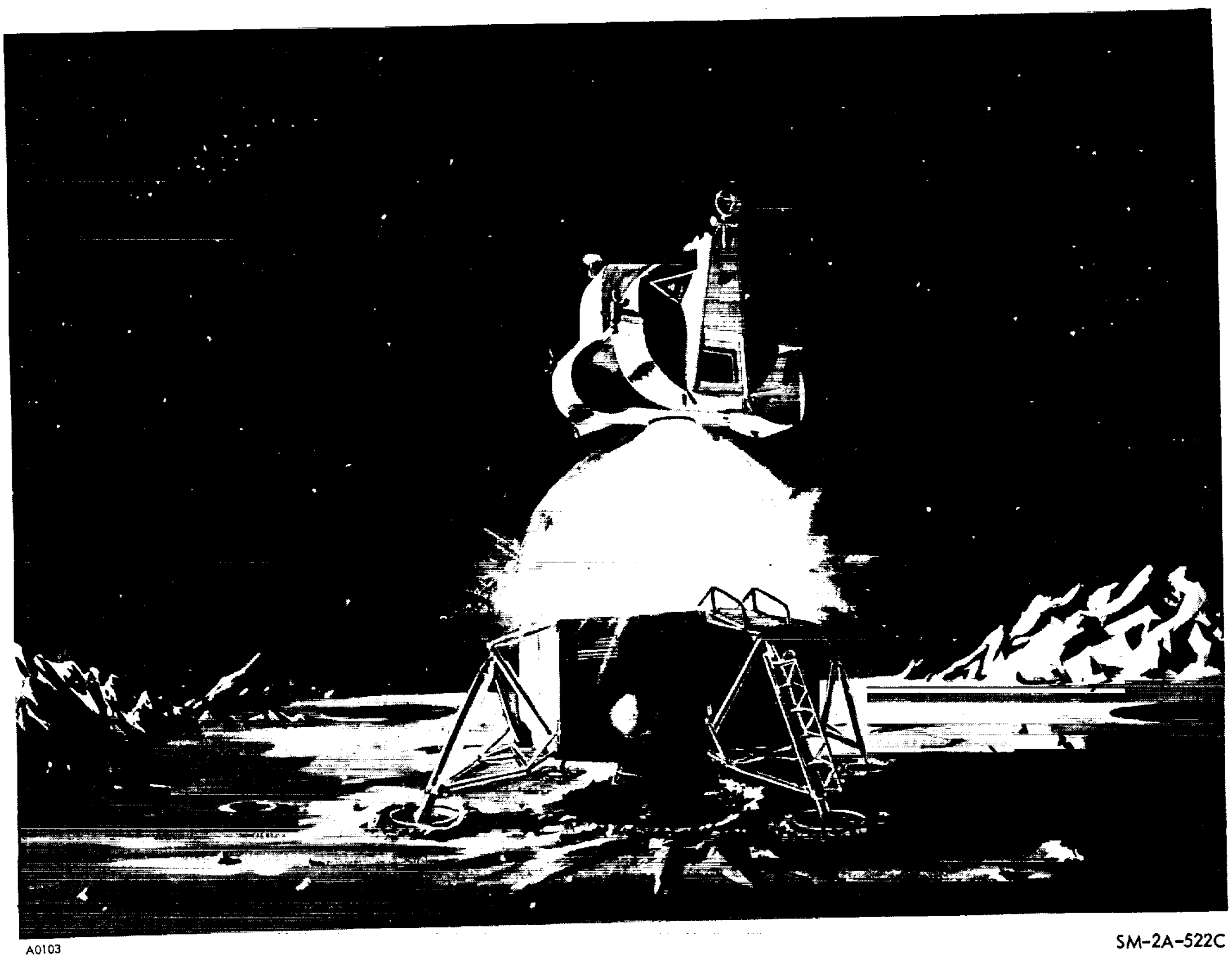

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Figure 8-16. Lunar Ascent

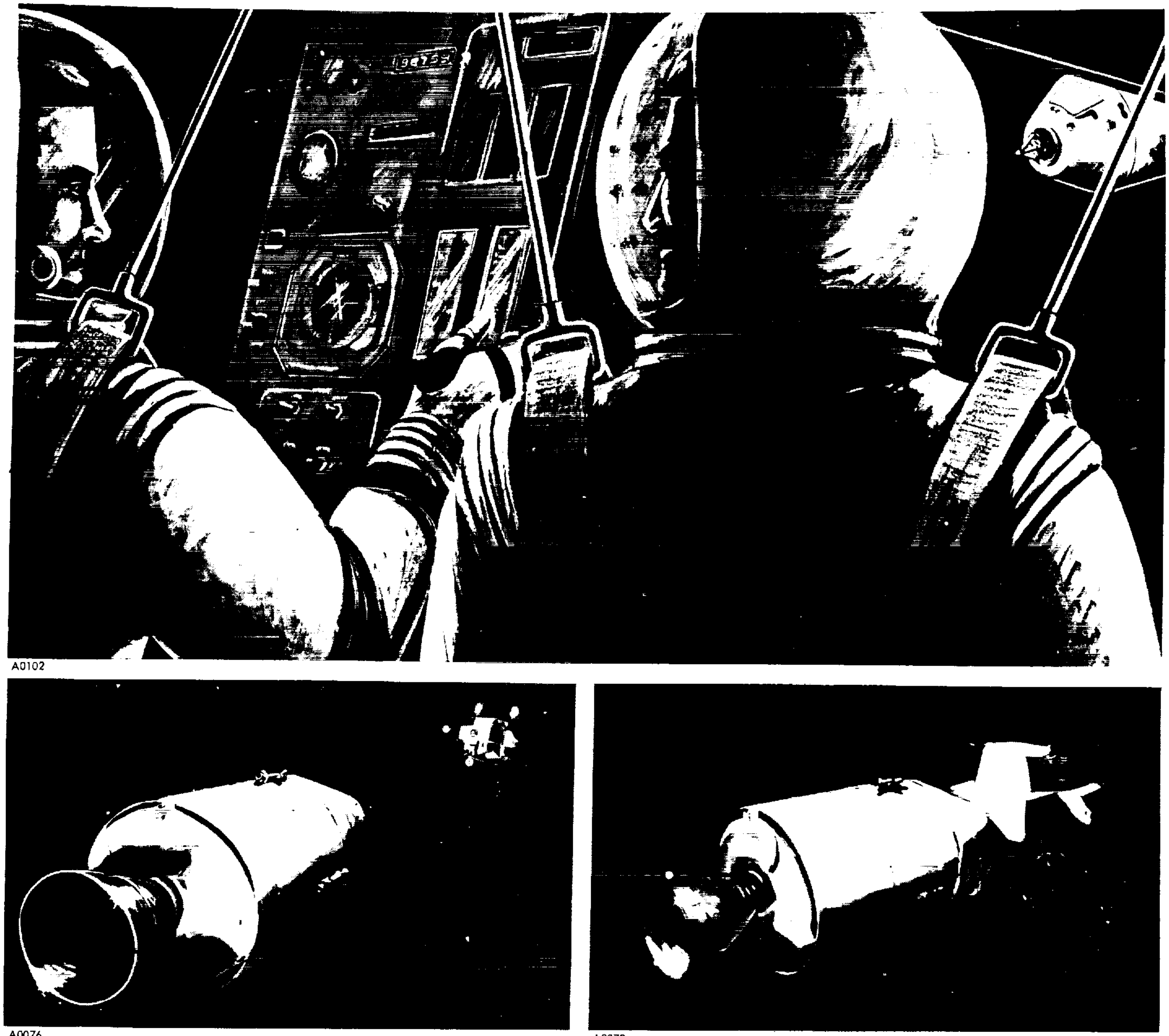

Figure 8-17. Rendezvous

8-78. RENDEZVOUS.

8-79. The rendezvous operations begin during the LEM ascent phase. The ascent trajectory may require up to three midcourse corrections to reach a collision course with the CSM. These corrections will be made with either the LEM ascent engine or the reaction control system. The final rendezvous maneuvers will include three terminal homing thrusts from the LEM reaction control system to reduce the relative velocity to a minimum.

8-80. The LEM crew will manually control the lunar excursion module within a range of approximately 500 feet from the spacecraft, with a relative velocity of 5 feet per second, or less. Both the spacecraft and the LEM are capable of performing the final rendezvous and docking maneuvers required. The spacecraft is normally stabilized in passive mode with the lunar excursion module operationally active to effect rendezvous.

8-81. The final rendezvous maneuvers of the LEM to effect contact with the spacecraft, are performed using the reaction control engines. Docking alignment and closing velocity will be verified and necessary manual operational control established to effect engagement of the drogue and probe. Following verification of the drogue and probe engagement the latching sequence will be performed. Completion of final docking will be verified by both the LEM and spacecraft crewmembers. The postdocking status of the system will be determined and, when the docking maneuver is completed, information will be transmitted to manned space-flight network.

8-82. Following engagement of the drogue and probe, and four semi-automatic latches (initial docking), a LEM crewmember will remove the LEM upper hatch and secure the LEM to the CSM by engaging the remaining 8 of the 12 latches provided (final docking). (See figure 3-22.) After the drogue and probe are removed and stowed in the LEM, the pressures are allowed to equalize and the C/M access hatch is removed.

8-83. Following completion of final docking, the LEM systems will be secured. The LEM crew will then transfer the lunar scientific equipment and samples to the command module and store them. The two crewmembers will enter the command module and secure the access hatch between the command module and the lunar excursion module.

8-84. A system status check will be made of the spacecraft operational systems and the status for separation from the LEM will be communicated to MSFN. The sequence for separation will then be initiated. The LEM will be pyrotechnically released from the CSM and the service module reaction control engines activated to translate the spacecraft away from the lunar excursion module.

8-85. Following this separation, the spacecraft will be operationally maneuvered for the inertial measurement unit alignment. After fine-alignment of the inertial measurement unit, a series of three navigational sightings will be made to establish the transearth injection parameters. The transearth injection trajectory is computed by the Apollo guidance computer and subsequently confirmed with MSFN. The final transearth injection operational parameters, service module reaction control system ullage acceleration, required service propulsion system firing time, incremental velocity, thrust vector, and transearth injection attitude are determined and confirmed by MSFN.

8-86. TRANSEARTH INJECTION AND COAST.

8-87. The transearth injection phase covers the period of time the service propulsion system burns when injecting the spacecraft into a transearth trajectory.

8-88. For each lunar orbit there exists one opportunity for transearth injection. Injection occurs behind the moon, with respect to the earth, nominally one orbit after completion of rendezvous with the lunar excursion module. The service propulsion system ullage acceleration is manually initiated to begin the earth injection sequence. The injection velocity increment for the predetermined transit time to earth is initiated with the service propulsion system thrust controlled by the guidance and navigation system.

8-89. The transearth coast phase begins with service propulsion system engine cutoff following transearth injection and ends at the entry interface altitude of 400,000 feet.

8-90. The transearth injection will be operationally performed to place the spacecraft in a return trajectory toward the earth, and will require a minimum of operational maneuvers and corrections. A ΔV budget sufficient to provide a total velocity correction of 100 feet per second during the transearth coast phase is provided. Nominally, three transearth velocity corrections may be made: one near the moon, the second approximately near the midpoint of the return trajectory, and the third ΔV increment near the earth.

9-91. The primary operations which occur consist of periodic systems checks, trajectory verification, determination of the ΔV corrections required, preparation for jettison of the service module, service module jettison from the command module, and preparations for earth entry.

8-92. The midcourse corrections are determined by means of sequential trajectory verifications. The Apollo guidance computer computes variations from the required trajectory parameters; and determines velocity changes (if necessary), thrust vectors, and firing time. This data is confirmed with MSFN. The ΔV is operationally implemented and the verification of the velocity correction is subsequently determined after each midcourse correction.

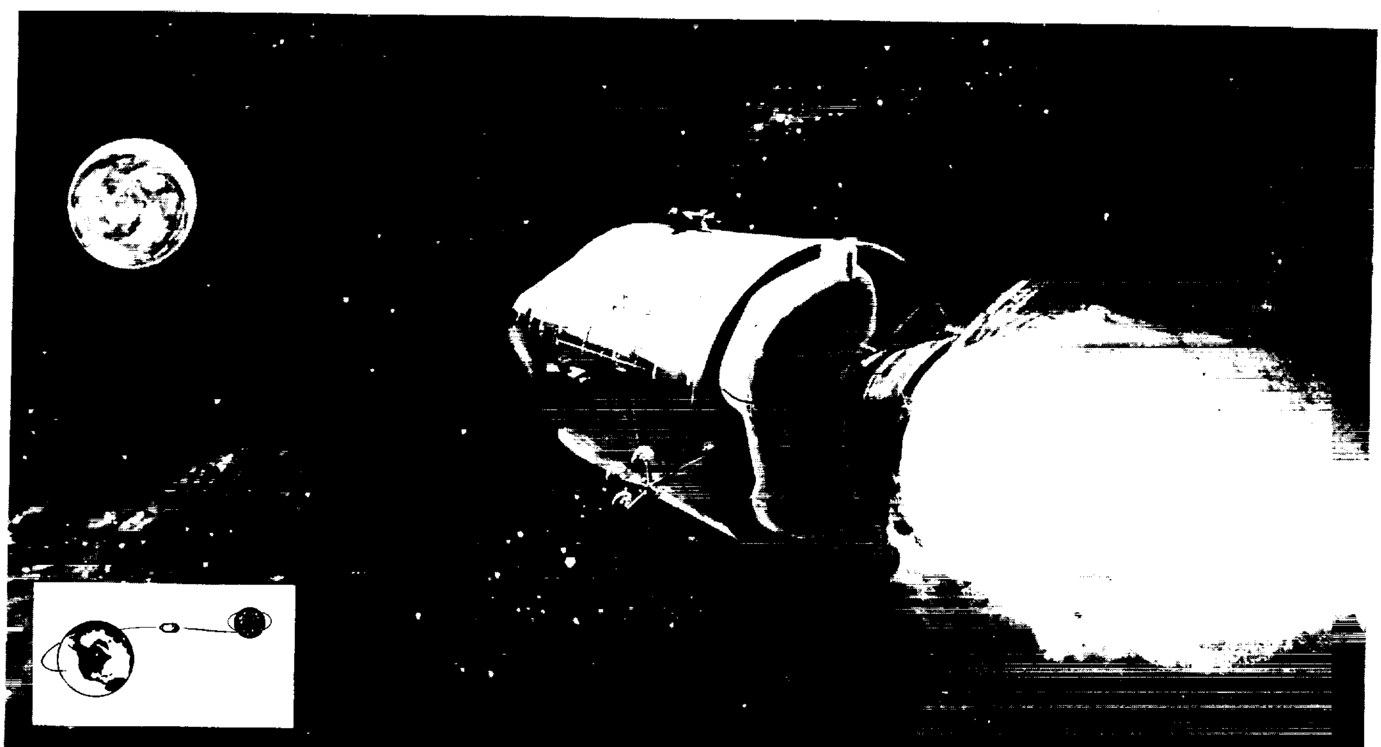

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Figure 8-18. Transearth Injection and Coast

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Figure 8-19. Service Module Jettison

8-93. SERVICE MODULE JETTISON.

8-94. Following the last midcourse correction, preparatory activity will be initiated for jettisoning the service module. The near-earth entry corridor and pre-entry parameters for separation of the service module from the command module will be determined by MSFN and confirmed by the Apollo guidance computer. The final systems check will be performed and the spacecraft oriented for service module jettison. The command module entry batteries will be activated, and the service module operationally jettisoned by pyrotechnic separation of the adapter and subsequent translational thrust by the service module reaction control engines to effect space separation of the command module and service module. The command module is then oriented into an MSFN-confirmed entry attitude by the command module reaction control engines.

8-95. A status check will be made of the systems after service module separation. Final operational checks will be made of the systems for entry. Confirmation of the entry parameters will be made with MSFN. The entry monitor control and display will be operationally activated, entry alignment of the inertial measurement unit made, and utilization of the flight director attitude indicator and Apollo guidance computer implemented.

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Figure 8-20. Earth Entry

8-96. EARTH ENTRY.

8-97. The earth entry phase begins at an altitude of 400,000 feet and ends upon activation of the earth landing system.

8-98. The operational control of the entry is dependent on the range required from the 400,000-foot entry point to the landing area. For short-entry ranges, no skip-out maneuver is required. For entry ranges approaching the upper limit, a skip-out maneuver is required to attain the greater distance. In either case, the lift maneuver is controlled by rolling the command module using the reaction control engines. Operational control is normally maintained through the guidance and navigation system with the pilot providing a backup capability using the entry monitor display.

8-99. Entry into the earth atmosphere is sensed by a 0.05G signal indication. The entry attitude is determined on the flight director attitude indicator, and the entry monitor control display is observed. The Apollo guidance computer computes the range to "go" and provides navigation from the 0.05G point and time.

8-100. The range control maneuver is initiated by reaction control engines roll control and necessary pitch and yaw damping. The entry monitor display indicates the ΔV and G-level, the command module ΔV and G-level time history, and the survival display requirements. The guidance and navigation system executes the required reaction control system roll commands.

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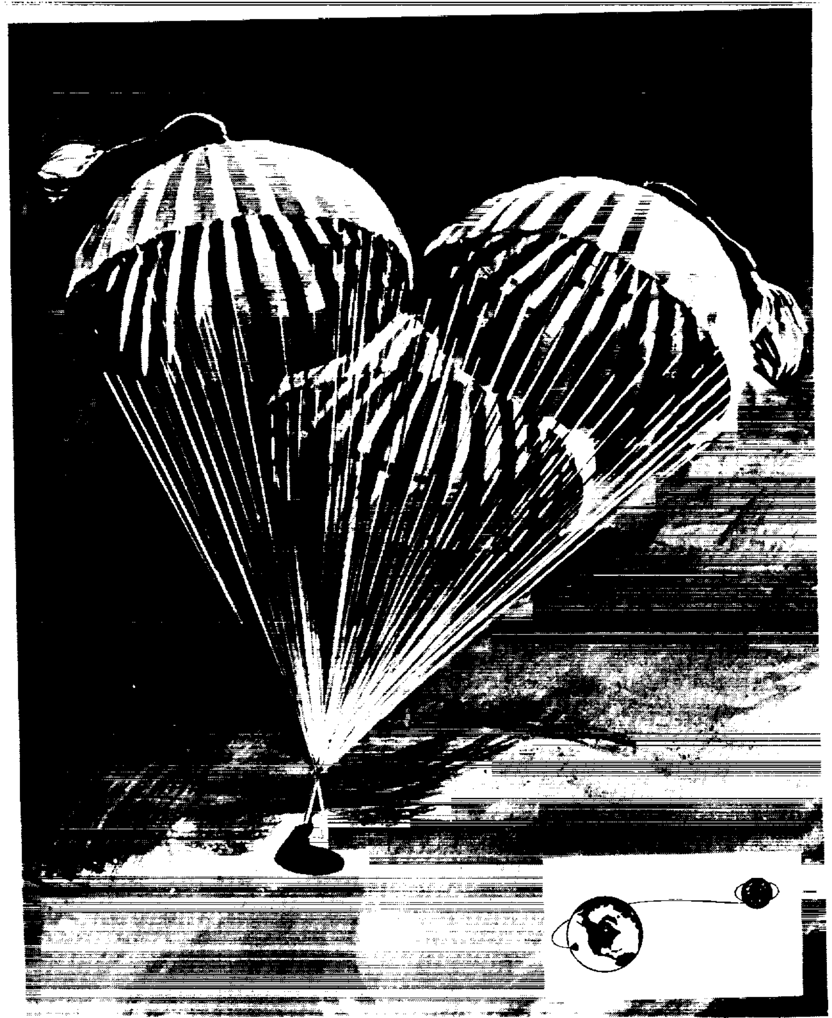

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Figure 8-21. Earth Landing

8-101. EARTH LANDING.

8-102. The earth landing phase begins when the earth landing system is operationally armed at an altitude of 100,000 feet and ends with touchdown. The earth landing system automatic sequencer pyrotechnically ejects the forward heat shield at approximately 24,000 feet. Two seconds later, the two drogue parachutes are mortar-deployed in a reefed condition. The reefing lines are pyrotechnically severed and the drogue parachutes open fully in approximately 9 seconds to orient the C/M apex upward during descent to 10,000 feet. Three pilot parachutes are automatically mortar-deployed at 10,000 feet, and the drogue parachutes are pyrotechnically disconnected. The pilot parachutes in turn, deploy the three main parachutes to a line-stretch, reefed condition. The reefing lines are pyrotechnically severed and the main parachutes open fully in approximately 10 seconds. The main parachutes lower the command module to touchdown and impact at a terminal descent velocity, assuring an impact G-level consistent with the safety of the crew. The main parachute attach lines are pyrotechnically severed upon touchdown.

8-103. During the final part of the main parachute descent, the recovery communication systems are activated and transmit a location signal for reception by the operational recovery forces.

Figure 8-22. Recovery Operations, Primary Landing

8-104. RECOVERY OPERATIONS.

8-105. The recovery operations phase begins with touchdown and ends with the recovery of the crew and retrieval of the command module. The VHF communication system is deployed and begins transmitting a repeating location signal for reception by the recovery task forces deployed in the area of predicted touchdown. Voice communication capability is also provided by an H-F communications systems.

8-106. If touchdown occurs in water (primary landing), fluorescent dye goes into solution, coloring the water a bright, fluorescent, yellow-green over an extended area and lasting approximately 12 hours. The dye should be visible to recovery force aircraft or ships for a considerable distance. A flashing beacon light is also provided for use at night.

8-107. Immediately following a water landing, after the main parachutes are pyrotechnically cut from the command module, the crew will assess the flotation status and capability of the command module. If the command module is in the inverted (stable II) flotation attitude, the crew will activate the uprighting system. When the command module achieves an upright (stable I) flotation attitude, the crew will remain in the command module or, if necessary, will leave in the inflatable liferaft provided for the three crewmembers. Steps will be taken, as necessary, to effectively secure the command module for optimum flotation stability and subsequent retrieval. The capability is to be provided for helicopter pickup of the command module, using the recovery pickup loop, or a nearby ship may pick up the command module. The three crewmembers may be picked up by helicopter, ship, or boat. Land ground forces may pick up the command module if the land touchdown point is in an accessible area.

8-108. The flotation design will provide a survivable flotation capability for a minimum of 48 hours, under design sea conditions. A water landing provides fewer touchdown hazards and a correspondingly greater safety for the crew.

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Figure 8-23. Recovery Operations, Backup Landing

APOLLO SUPPORT MANUALS

A-1. GENERAL.

A-2. Apollo support manuals consist of published data packages to support the Apollo program. The manuals are categorized into general series and defined by specific letter/number combinations as follows:

• SM1A-1	Index of Apollo Support Manuals and Procedures
• SM2A-02	Apollo Spacecraft Familiarization Manual
• SM2A-03-(S/C No.)	Preliminary Apollo Operations Handbook, Command and Service Module
• SM2A-03A-(S/C No.)	Preliminary Apollo Operations Handbook, Command and Service Module (Confidential Supplement to SM2A-03)
• SM2A-08-(S/C No.)	Apollo Recovery and Postlanding Operations Handbook
• SM3A-200	Apollo Ground Support Equipment Catalog
• SM6A-(Series No.)	Apollo Training Equipment Maintenance Handbooks
-22	Electrical Power System Trainer
-23	Environmental Control System Trainer
-24	Stabilization Control System Trainer
-25	Sequential Flow System Trainer
-26	Propulsion System Trainer
-41-1 and -2	Mission Simulators Maintenance and Operations Manual
• SM6T-2-02	Apollo Mission Simulator Instructor Handbook

A-3. INDEX OF APOLLO SUPPORT MANUALS AND PROCEDURES.

A-4. Index SM1A-1 is published periodically and provides a listing of all Apollo Support Manuals and Procedures in publication.

A-5. APOLLO SPACECRAFT FAMILIARIZATION MANUAL.

A-6. The familiarization manual (SM2A-02) presents a general, overall description of the Apollo program. Coverage includes physical configuration, functional operation, the test program, and the missions of the equipment utilized within the scope of the Apollo program. General terms are used in the descriptive text with sufficient detail to ensure comprehension.

A-7. PRELIMINARY APOLLO OPERATIONS HANDBOOK, COMMAND AND SERVICE MODULE.

A-8. A preliminary Apollo operations handbook (SM2A-03 and -03A) is a preliminary version of the Apollo Operations Handbook designed as a single source of spacecraft data and operational procedures. The handbook provides detailed spacecraft operating instructions and procedures for use by the crew during all phases of the latest, manned Block I mission. This involves normal, alternate, backup, malfunction, and contingency procedures, crew procedures, and checklists. CSM and LEM interface instructions and procedures applicable to the command module are included, with information on spacecraft controls and displays, systems data, crew personal equipment, in-flight experiments, and scientific equipment.

A-9. APOLLO RECOVERY AND POSTLANDING OPERATIONS HANDBOOK.

A-10. The Apollo recovery and postlanding operations handbook (SM2A-08-S/C No.) provides detailed information for retrieval of the recoverable portion of the spacecraft. Descriptive text and illustrations specify procedures and provide the information necessary to perform recovery and postlanding operations of each manned and unmanned command module scheduled for recovery.

A-11. APOLLO GROUND SUPPORT EQUIPMENT CATALOG.

A-12. An Apollo ground support equipment catalog (SM3A-200) is provided to identify and illustrate items of specific auxiliary, checkout, handling, and servicing equipment associated with Project Apollo.

A-13. APOLLO TRAINING EQUIPMENT MAINTENANCE HANDBOOKS.

A-14. An Apollo training equipment maintenance handbook (SM6A-series) is provided for the maintenance of the systems trainers and mission simulators.

A-15. APOLLO MISSION SIMULATOR INSTRUCTOR HANDBOOK.

A-16. An instructor's handbook (SM6T2-02) is provided for the mission simulators. The handbook provides the necessary information for training astronauts on the Apollo mission simulators.

GLOSSARY OF ABBREVIATIONS, SYMBOLS, AND TERMS

This glossary lists terminology found in Apollo documentation and engineering drawings. Frequently used common terms, which are industry standard, have been omitted for brevity. This glossary will be updated to reflect the latest changes during each revision of the manual.

ABBREVIATIONS

AAO	Astronauts Activities Office
ABD	Airborne Ballistics Division (NASA)
AC	Audio center
ACA	Associate contractor administration
ACE	Acceptance checkout equipment
ACF	American Car and Foundry
ACM	Audio center module
ACME	Attitude control and maneuvering electronics
A&CO	Assembly and checkout
ACR	Associate contractor
ACRC	Audio center - receiver
ACS	Attitude control and stabilization
ACSB	Apollo crew systems branch
ACSP	AC Spark Plug, Division of General Motors
ACTM	Audio center - transmitter
ACV	AC volts
AD	Apollo development
A/D	Analog-to-digital
ADA	Angular differentiating accelerometer
ADC	Analog-to-digital converter
ADF	Automatic direction finding (equip.)
ADP	Automatic data processing
AEB	Aft equipment bay
AEDC	Arnold Engineering Development Center
AERO	Aeroballistics (MSFC)
AERO-A	AERO - Aerodynamics analysis
AERO-D	AERO - Dynamics analysis
AERO-DIR	AERO - Director
AERO-E	AERO - Experimental aerodynamics
AERO-F	AERO - Flight evaluation
AERO-G	AERO - Aerophysics and astrophysics
AERO-P	AERO - Future projects
AERO-PCA	AERO - Program coordination and administration
AERO-PS	AERO - Projects staff
AERO-TS	AERO - Technical and scientific staff
AF	Audio frequency
AFCS	Automatic flight control system
AFETR	Air Force eastern test range
AGAA	Attitude gyro accelerometer assembly
AGANI	Apollo guidance and navigation information
AGAP	Attitude gyro accelerometer package (superseded by AGAA)
AGC	Apollo Guidance Computer (used by MIT)
AGC	Aerojet General Corporation
AGCS	Automatic Ground Control Station (NASA)
AGCU	Attitude gyro coupling unit

AGE	Apollo Guidance and Navigation Equipment (used by MIT)
AGE	Aerospace ground equipment
AIAA	American Institute of Aeronautics & Astronautics
AIDE	Aerospace installation diagnostic equipment
AII	Apollo implementing instructions (NAA, S&ID)
ALFA	Air lubricated free attitude
ALIAS	Algebraic logic investigations of Apollo systems
AM	Amplitude modulation
AMG	Angle of middle gimbal
AMMP	Apollo master measurements program
AMOO	Aerospace Medical Operations Office (MSC)
AMPTF	Apollo Mission Planning Task Force
AMR	Atlantic Missile Range (Superseded by ETR)
AMRO	Atlantic Missile Range Operations (MSC)
AMS	Acoustic measurement system
AMS	Apollo mission simulator
AMW	Angular momentum wheel
AOH-CSM	Apollo Operations Handbook-CSM
AOH-LEM	Apollo Operations Handbook-LEM
AORA	Atlantic Ocean Recovery Area
AOS	Atlantic Ocean ship
AP	Access point
AP	Accelerometer package
APC	Procurement and Contracts Division (MSC)
APCA	Procurement Apollo
APCAL	Procurement Apollo lunar excursion module
APCAN	Procurement Apollo navigation and guidance
APCAS	Procurement Apollo spacecraft
APCAT	Procurement Apollo test and instrumentation
APCR	Apollo program control room
APD	Advanced program development

APHFFF	Ames Prototype Hypersonic Free Flight Facility
APK	Accelerometer package
APO	Apollo project office
APP	Access point PACE
APTT	Apollo Part Task Trainer
APU	Auxiliary power unit
AQ	Apollo qualification
ARA	Auxiliary recovery antenna
ARC	Ames Research Center (NASA) (Moffet Field, Calif.)
ARE	Apollo reliability engineering
AREE	Apollo reliability engineering electronics
ARIS	Advanced range instrumentation ships
ARM	Apollo Requirements Manual
ARS	Attitude reference system
ASCS	Automatic stabilization and control system
ASDD	Apollo signal definition document
ASDTP	Apollo Spacecraft Development Test Plan
ASFTS	Auxiliary systems function test stand
AS/GPD	Attitude set and gimbal position display
ASI	Apollo systems integration
ASM	Apollo Systems Manual
ASP	Apollo spacecraft project
ASPI	Apollo supplemental procedural information
ASPO	Apollo Spacecraft Project Office
ASTR	Astronics (MSFC)
ASTR-A	ASTR - Advanced studies
ASTR-ADM	ASTR - Administrative
ASTR-E	ASTR - Electrical systems integration
ASTR-F	ASTR - Flight dynamics
ASTR-G	ASTR - Gyro and stabilizer
ASTR-I	ASTR - Instrumentation development
ASTR-M	ASTR - Electromechanical engineering
ASTR-N	ASTR - Guidance and control systems

ASTR-P	ASTR - Pilot manufacturing development
ASTR-PC	ASTR - Program coordination
ASTR-R	ASTR - Applied research
ASTR-TSA	ASTR - Advanced research and technology
ASTR-TSJ	ASTR - Saturn
ASTR-TSR	ASTR - Reliability
ATC	Assistant test conductor
ATO	Apollo Test and Operations
AT&O	Apollo Test and Operations (ATO is preferred)
ATR	Apollo test requirement
ATS	Atlantic tracking ship
AUTO	Automatic
A-V	Audio-visual
AVC	Automatic volume control
AVSS	Apollo Vehicle Systems Section (NASA)
AWI	Accommodation weight investigation
BAC	Boeing Aircraft Company
BATT	Battery
BCD	Binary coded decimal
BCO	Booster engine cutoff
BDA	Bermuda (remote site)
BECO	Booster engine cutoff
BER	Bit error rate
BG	Background
BLWR	Blower
BM	Bench maintenance
B/M	Bench maintenance (BM is preferred)
BMAG	Body-mounted attitude gyro
BME	Bench maintenance equipment
BMG	Body-mounted (attitude) gyro (BMAG is preferred)
BOA	Broad ocean area
BOD	Beneficial occupancy data
BP	Blood pressure
BP	Boilerplate
BPC	Boost protective cover
BPS	Bits per second
BSI	Booster situation indicator
B/U	Backup
CAS	Crewman alignment sight
C/B	Circuit breaker
CBA	C-band transponder antenna
CBC	Complete blood count
CBX	C-band transponder
cc	Cubic centimeter
CCTV	Closed-circuit television
CCW	Counterclockwise
C/D	Countdown
C&D	Communication and data
CDC	Computer Development Center (NASA)
CDCM	Coupling display manual control - IMU
CDCO	Coupling display manual control - optics
CDOH	Coupling display optical hand controller
CDR	Critical design review
CDRD	Computations and Data Reduction Division (MSC)
CDSC	Coupling display SCT manual control
C&DSS	Communications and data subsystems
CDU	Coupling display unit
CDU	Coupling data unit
CDUM	Coupling display unit - IMU
CDUO	Coupling display unit - optics
CEPS	Command module electrical power system
C/F	Center frequency
CFAE	Contractor-furnished airborne equipment
CFD	Cumulative frequency distribution
CFE	Contractor-furnished equipment
CFDF	Crew flight data file
CFM	Cubic feet per minute
cg	Center of gravity
CGSS	Cryogenic gas storage system
CH_4	Methane
CHGE	Charger
C&I	Communication and Instrumentation
CIF	Central Information Facility (AMR)
CIR&SEP	H_2 Circulation, water separation centrifuge, and glycol circulation
CIS	Communication and instrumentation system

C&IS	Communication and instrumentation system (CIS is preferred)
CL	Closed-loop
CLM	Circumlunar mission
C/M	Command module
CMM	Communications and Telemetry (used by MIT)
CO	Carbon monoxide
C/O	Checkout
C/O	Cutoff
CO_2	Carbon dioxide
COMP	Compressor
CP	Control panel
CP	Control Programer
CPE	Chief project engineer
CPEO	CPE Engineering Order
CPO	Central Planning Office (MSFC)
cps	Cycles per second
CPS	Critical path schedule
CRT	Cathode-ray tube
CRYO	Cryogenics
CS	Communication system
CSD	Computer systems director
CSD	Crew System Division (MSC)
CSM	Command and service module
CSS	Crew safety system
CSS	Cryogenic storage system
CST	Combined systems test
CSTU	Combined systems test unit
CTE	Central timing equipment
CTL	Component Test Laboratory (NASA)
CTN	Canton Island (remote site)
CTU	Central timing unit
CUE	Command uplink electronics (PACE)
CW	Clockwise
CW	Continuous wave
CWG	Constant-wear garment
CYI	Canary Islands
DA	Dip angle
DA	Double amplitude
DAC	Digital-to-analog converter
DAE	Data acquisition equipment
DART	Director and response tester
DAS	Data acquisition system
db	Decibel
DBM	Decibels with respect to one milliwatt
DBW	Decibels with respect to one watt
D&C	Displays and controls
DCA	Design change authorization
DCCU	Decommutator conditioning unit (PACE)
DCIB	Data communication input buffer
DCOS	Data communication output selector
DCS	Design control specification
DCU	Display and control unit
DCV	DC volts
DDP	Data distribution panel
DDS	Data display system
DE	Display electronics
DEA	Display electronics assemblies
DECA	Display/AGAP electronic control assembly
DEI	Design engineering inspection
DF	Direction finding
D/F	Direction finder
DFS	Dynamic flight simulator
DIM	Design information manual
DISC	Discharge
DISPLAY/ AGAA ECA	Display and attitude gyro, accelerometer assembly-electronic control assembly
DM	Design manual
DNR	Downrange
DOD	Department of Defense
DOF	Degree of freedom
DOF	Direction of flight
DOVAP	Doppler velocity and position
DP	Design proof
DPC	Data processing center
DPDT	Double-pole double-throw
DPST	Double-pole single-throw
DRM	Drawing requirements manual
DSB	Double sideband
DSE	Data storage equipment
DSIF	Deep-space instrumentation facility
DSKY	Display and keyboard
DTCS	Digital test command system (PAGE)

DTS	Data transmission system	ELS	Earth landing system
DTVC	Digital transmission and verification converter (PACE)	E/M	Escape motor
		EMD	Entry monitor display
DVD	Delta velocity display	EMG	Electromyograph, electromyography, electromyogram
DVO	Delta velocity on/off		
DVU	Delta velocity ullage	EMI	Electromagnetic interference
		EMS	Entry monitor system
EBW	Explosive bridgewire	ENVR	Environmental
ECA	Electronic control assembly	EO	Earth orbit
ECA	Engineering change analysis	EO	Engineering order
ECAR	Electronic control assembly - roll	E&O	Engineering and operations (building)
ECD	Engineering control drawing	EOD	Explosive ordnance disposal
ECD	Entry corridor display	EOL	Earth orbit launch
ECET	Electronic control assembly - engine thrust	EOM	Earth orbital mission
		EOR	Earth orbital rendezvous
ECG	Electrocardiograph, electrocardiography, electrocardiogram	EPDS	Electrical power distribution system
		EPS	Electrical power system
ECK	Emergency communications key	EPSTF	Electrical power system test facility
ECN	Engineering change notice		
ECO	Engine cutoff	EPUT	Events per unit time
ECO	Engineering Change Order (MSC)	ERG	Electroretinograph, electroretinography, electroretinogram
ECPY	Electronic control assembly - pitch & yaw		
		ERP	Eye reference point
ECS	Environmental control system	ERS	Earth recovery system
ECU	Environmental control unit	ERU	Earth rate unit (15 degrees/hour)
EDL	Engineering development laboratories (NAA, S&ID)		
		ESB	Electrical Systems Branch (MSC)
EDP	Electronic data processing		
EDPM	Electronic data processing machine	ESE	Engineering support equipment
EDS	Emergency detection system	ESS	Emergency Survival System (NASA)
EED	Electroexplosive device		
EEG	Electroencephalograph, electroencephalography, electroencephalogram	ESS	Entry survival system
		ESTF	Electronic System Test Facility (NASA)
EET	Equivalent exposure time	ESV	Emergency shutoff valve
EFSSS	Engine failure sensing and shutdown system	ET	Escape tower
		E/T	Escape tower
EHF	Extremely high frequency	ETF	Eglin Test Facility
EI	Electromagnetic interference	ETOC	Estimated time of correction
EI	Electronic interface	ETR	Eastern test range
EKG	Electrocardiograph, electrocardiography, electrocardiogram	EVAP	Evaporator
		EVT	Extravehicular transfer
ELCA	Earth landing control area	FACT	Flight acceptance composite test

FAE	Final approach equipment
FAP	Fortran assembly program
FAX	Facsimile transmission
FC	Ferrite core
F/C	Fuel cell
F/C	Flight control
FCD	Flight control division
FCH	Flight controller's handbook
FCOB	Flight Crew Operations Branch (NASA)
FCOD	Flight Crew Operations Division (MSC)
FCSD	Flight Crew Support Division (MSC)
FCT	Flight Crew Trainer (IMCC)
FD	Flight Director (NASA)
FDAI	Flight director attitude indicator
FDO	Flight dynamics officer
FDRI	Flight director rate indicator
FEB	Forward equipment bay
FEO	Field engineering order
FF	Florida Facility
FHS	Forward heat shield
FIDO	Flight dynamics officer
FLSC	Flexible linear-shaped charge
FM	Frequency modulation
FMA	Failure mode analysis
FMD&C	Flight mechanics, dynamics and control
FMX	FM transmitter
FO	Florida Operations
FOD	Flight Operations Division (MSC)
FOF	Flight operations facilities
FORTRAN	Formula translation
FOS	Flight operations support
FP	Fuel pressure
FPO	Future Projects Office (MSFC)
FPS	Frames per second
fps	Feet per second
FQ	Flight qualification
FQR	Flight qualification recorder
FRDI	Flight research and development instrumentation
FRF	Flight readiness firing
FSK	Frequency shift keying
FTP	Flight test procedure
GAEC	Grumman Aircraft Engineering Corp.
GC	Gigacycles (1000 megacycles)
G&C	Guidance and control
G&CEP	Guidance and control equipment performance
GCU	(Attitude) gyro coupling unit (AGCU is preferred)
GDC	Gyro display coupler
GETS	Ground equipment test set
GFAE	Government-furnished aeronautical equipment
GFE	Government-furnished equipment
GFP	Government-furnished property
GG	Gas generator
GH_2	Gaseous hydrogen
GHE	Gaseous helium
GHe	Gaseous helium (preferred)
GLY	Glycol
GMT	Greenwich Mean Time
G&N	Guidance and navigation
G&NS	Guidance and navigation system
GN_2	Gaseous nitrogen
GNC	Guidance and navigation computer
GNE	Guidance and navigation electronics
GO_2	Gaseous oxygen
GORP	Ground operational requirements plan
GOSS	Ground Operational Support System (superseded by MSFN)
GOX	Gaseous oxygen (superseded by GO_2)
GP	General purpose
GPD	Gimbal position display
GPI	Gimbal position indicator
gpm	Gallons per minute
GSDS	Goldstone duplicate standard (standard DSIF equipment)
GSE	Ground support equipment
GSFC	Goddard Space Flight Center (NASA) (Greenbelt, Md.)
GSP	Guidance signal processor
GSPO	Ground Systems Project Office (MSC)
GSP-R	Guidance signal processor - repeater
GSR	Galvanic skin response
GSSC	General Systems Simulation Center (NASA)
GSSC	Ground Support Simulation Computer (MCC)
GTI	Grand Turk Island
GTK	Grand Turk

GTP	General test plan
G vs T	Deceleration units of gravity versus time
G vs V	Deceleration units of gravity versus velocity
GYI	Grand Canary Island (remote site)
GYM	Guaymas, Mexico (remote site)
H_2	Hydrogen
H/A	Hazardous area
HAA	High altitude abort
HAW	Kauai Island, Hawaii (remote site)
HBW	Hot bridgewire
HC	Hand control
HE	Helium (He is preferred)
He	Helium
H/E	Heat exchanger
HF	High frequency
H/F	Human factors
HFA	High frequency recovery antenna
HFX	High frequency transceiver
HGA	High-gain antenna (2 KMC)
HGB	Hemoglobin
HI	High
HME	Office of Manned Space Flight (MSFC)
H_2O	Water
H_2S	Hydrogen sulfide
HS	Hot short
HS	Hydrogen sulfide
H/S	Heat shield
H-S	Hamilton Standard
H-S	Horizon scanner
HS/C	House spacecraft
HSD	High-speed data
HTRS	Heaters
HW	Hotwire
HWT	Hypersonic wind tunnel
H/X	Heat exchanger
IA	Input axis
IAD	Interface analysis document
IAS	Indicated air speed
IC	Intercommunication equipment
I/C	Intercom
ICA	Item change analysis
ICD	Interface control document
ICM	Instrumentation and Communications monitor
ID	Inside diameter
IESD	Instrumentation and Electronic Systems Division (MSC)
IF	Intermediate frequency
I/F	Interface
IFM	In-flight maintenance
IFT	In-flight test
IFTM	In-flight test and maintenance
IFTS	In-flight test system
IG	Inner gimbal
IGA	Inner gimbal axis
IL	Instrumentation Lab (MIT)
IL	Inertial Lab (NASA)
IL	Internal letter
ILCC	Integrated launch checkout and control
IMCC	Integrated Mission Control Center (superseded by MCC)
IMU	Inertial measurement unit
IND	Indicator
INS	Inertial navigation system
INV	Inverter
IORA	Indian Ocean Recovery Area
IOS	Indian Ocean ship (tracking)
I/P	Impact predictor
IR	Infra red
IRG	Inertial rate gyro
IRIG	Inertial rate integrating gyroscope (used by MIT)
IRP	Inertial reference package
IS	Instrumentation system
I_{sp}	Specific impulse
IST	Integrated systems test
IU	Instrumentation unit
I/U	Instrumentation unit (IU is preferred)
IUA	Inertial unit assembly
J/M	Jettison motor
KC	Kilocycle (1000 cycles per second)
KMC	Kilomegacycle
KNO	Kano, Nigeria (remote site)
KOH	Potassium hydroxide

KSC	Kennedy Space Center
KW	Kilowatt
LAC	Lockheed Aircraft Corporation
LAET	Limiting actual exposure time
LC	Launch complex
LC-39	Launch complex 39
LCC	Launch Control Center (MCC)
LCE	Launch complex engineer
LCS	Launch control system
L/D	Lift-drag ratio
LDGE	LEM dummy guidance equipment
LDP	Local data package
LDT	Level detector
LE	Launch escape
L/E	Launch escape (LE is preferred)
LEB	Lower equipment bay
LEC	Launch escape control
LECA	Launch escape control area
LEM	Launch escape motor
LEM	Lunar excursion module
LES	Launch escape system
LESC	Launch escape system control
LET	Launch escape tower
LEV	Launch escape vehicle
LGC	LEM guidance computer
LGE	LEM guidance equipment
LH	Left-hand
LH_2	Liquid hydrogen
LHA	Local hour angle
LHE	Liquid helium
LHe	Liquid helium (preferred)
LHFEB	Left-hand forward equipment bay
LHSC	Left-hand side console
LiOH	Lithium hydroxide
LJ	Little Joe
LL	Low-level
LLM	Lunar landing mission
LLM	Lunar landing module
LLOS	Landmark line of sight
LLV	Lunar landing vehicle
LM	Landmark
L&M	Light and Medium Vehicles Office (MSFC)
LMSC	Lockheed Missile and Space Company
LN_2	Liquid nitrogen
LO	Launch operations
LO	Low
L/O	Lift-off
LO_2	Liquid oxygen
LOC	Launch Operations Center (NASA)(Cocoa Beach, Fla.)
LOD	Launch operations directorate
LOD	Launch Operations Division (NASA)(superseded by LOC)
LOM	Lunar orbital mission
LOR	Lunar orbital rendezvous
LOS	Line of sight
LOS	Loss of signal
LOX	Liquid oxygen (superseded by LO_2)
LP	Lower panel
LPA	Log periodic antenna
LPC	Lockheed Propulsion Company
LPGE	LEM partial guidance equipment
LRC	Langley Research Center (NASA)(Hampton, Va.)
LRC	Lewis Research Center (NASA)(Cleveland, Ohio)
LRD	Launch Recovery Division,
LSB	Lower sideband
LSC	Linear-shaped charge
LSD	Low-speed data
LSD	Life Systems Division (MSC) (superseded by CSD)
LSD	Launch systems data
LSS	Life support system
LTA	LEM test article
LTC	Launch vehicle test conductor
LTDT	Langley transonic dynamics tunnel
LUPWT	Langley unitary plan wind tunnel
LUT	Launch Umbilical Tower
LV	Launch vehicle
LV	Local vertical
L/V	Launch vehicle
LVO	Launch Vehicle Operations (MSFC)
LVOD	Launch Vehicle Operations Division (MSFC)
LVSG	Launch vehicle study group

MAN	Manual
MASTIF	Multi-axis spin test inertial facility
M.C.&W.S.	Master caution and warning system
MCA	Main console assembly
MCC	Mission control center
MCOP	Mission control operations panel
MCR	Master change record
MD	Master dimension
MDC	Main display console
MDF	Main distribution frame (MCC)
MDF	Mild detonating fuse
MDR	Mission data reduction
MDS	Malfunction detection system
MDS	Master development schedule
MDSS	Mission data support system
MDT	Mean downtime
MDV	Map and data viewer
MEC	Manual emergency controls
MEE	Mission essential equipment
MERu	Milli-earth rate unit (0.015 degree/hour)
MESC	Master event sequence controller
MEV	Million electron volts
MFG	Major functional group
MFV	Main fuel valve
MG	Middle gimbal
MG	Motor-generator
MGA	Middle gimbal axis
MGE	Maintenance ground equipment
MI	Minimum impulser
MIG	Metal inert gas
MIIA	Merrit Island Industrial Area
MIL	Miliradian
MILA	Merritt Island Launch Area (superseded by KSC)
MILPAS	Miscellaneous information listing program Apollo spacecraft
MIT	Massachusetts Institute of Technology
ML	Mold line
MLT	Mission life test
M/M	Maximum and minimum
MMH	Monomethylhydrazine (fuel)
MMHg	Millimeters of mercury
MMU	Midcourse measurement unit
MN A	Main bus A
MN B	Main bus B
MNEE	Mission nonessential equipment
M&O	Maintenance and operation
MOC	Master operations control
MOCR	Mission Operations Control Room (MCC)
MODS	Manned Orbital Development Station (MSC)
MORL	Manned orbiting research laboratory
MOV	Main oxidizer valve
MPTS	Multipurpose tool set
MRCR	Measurement requirement change request
MRO	Maintenance, repair, and operation
M&S	Mapping and surveying
MSC	Manned Spacecraft Center (NASA) (Clear Lake, Texas)
MSC-FO	Manned Spacecraft Center - Florida operations
MSD	Mission systems data
MSFC	Marshall Space Flight Center (NASA) (Huntsville, Ala.)
MSFC-LVO	Marshall Space Flight Center - Launch Vehicle Operations
MSFN	Manned Space Flight Network (formerly GOSS)
MSFP	Manned space flight program
MT	Magnetic tape
MTF	Mississippi Test Facility (NASA)
MTS	Master timing system
MTU	Magnetic tape unit
MTVC	Manual thrust vector control
MU	Mockup
M/U	Mockup (MU is preferred)
MUC	Muchea, Australia (remote site)
MV	Millivolt
MVD	Map and visual display (unit)
MW	Milliwatt
MWP	Maximum working pressure
N_2	Nitrogen
NAA	North American Aviation
NAACD	NAA, Columbus Division
NAARD	NAA, Rocketdyne Division
NAASD	NAA, Space Division

NASA	National Aeronautics and Space Administration
N/B	Narrow band
NC	Nose cone
N/C	Normally closed
N&G	Navigation and guidance (G&N is preferred)
NH_4	Ammonium
N_2H_4	Hydrazine (fuel)
NM	Nautical mile
NMO	Normal manual operation
N/O	Normally open
NOS ESS	Nonessential
N_2O_4	Nitrogen tetroxide (oxidizer)
NPC	NASA procurement circular
NPDS	Nuclear particle detection system
NPSH	Net positive suction head
NRD	National Range Division
NRZ	Nonreturn to zero
NSC	Navigational star catalog
NSIF	Near Space Instrumentation Facility
NSM	Network status monitor
NST	Network support team
NTO	Nitrogen tetroxide (oxidizer)
NVB	Navigational base
O_2	Oxygen
OA	Output axis
OA	Ominiantenna
OAM	Office of Aerospace Medicine (NASA)
OAO	Orbital astronomical observatory
O&C	Operation and checkout
OCC	Operational control center
OCDU	Optics coupling display unit (G&N)
O&C/O	Operation and checkout (O&C is preferred)
OD	Operations directive
OD	Outside diameter (overall diameter)
ODDA	Office of Deputy Director for Administration (MSFC)
ODDRD	Office of Deputy Director for Research and Development (MSFC)

ODM	One-day mission
O/F	Orbital flight
O/F	Oxidizer-to-fuel ratio
OFB	Operational Facilities Branch (NASA)
OFO	Office of Flight Operations (NASA)
OG	Outer gimbal
OGA	Outer gimbal axis
OIB	Operations Integration Branch (NASA)
OIS	Operational instrumentation system (MCC)
OL	Overload
OL	Open-loop
OMSF	Office of Manned Space Flight (NASA)
OMU	Optical measuring unit
OOA	Open ocean area
OPS	Operations Director (NASA)
OR	Operations requirements document (range user)
OSS	Office of Space Sciences
OTDA	Office of Tracking and Data Acquisition
OTP	Operational test procedure
OTU	Operating test unit
OVERS	Orbital vehicle re-entry simulator
OXID	Oxidizer
ΔP	Pressure change
PA	Precision angle (used by MIT)
PA	Power amplifier
PA	Pad abort (used by NASA)
P/A	Pressure actuated
PACE	Prelaunch automatic checkout equipment (formerly SPACE)
PAFB	Patrick Air Force Base
PAM	Pulse-amplitude modulation
PATH	Performance analysis and test histories
P&C	Procurement and Contracts (MSFC)
P/C	Pitch control
PCCP	Preliminary contract change proposal
PCD	Procurement control document
PCM	Pitch control motor

PCM	Pulse-code modulation
PCME	Pulse-code modulation event
PCPL	Proposed change point line
PDA	Precision drive axis
PDD	Premodulation processor - deep-space data
PDU	Pressure distribution unit
PDV	Premodulation processor - deep-space voice
PE	Positive expulsion
PE	Project engineer
PEP	Peak envelope power
PERT	Program evaluation and review technique
PF	Preflight
PFL	Propulsion Field Laboratory (Rocketdyne)
PFM	Pulse-frequency modulation
PFRT	Preliminary Flight Rating Test
PGA	Pressure garment assembly
PGNCS	Primary guidance navigation control system
pH	Alkalinity to acidity content (hydrogen ion concentration)
PIAPACS	Psychophysical information acquisition processing and control system
PIGA	Pendulous integrating gyroscopic accelerometer
PIO	Public Information Office (MSFC)
PIP	Pulsed integrating pendulous (accelerometer)
PIPA	Pulsed integrating pendulous accelerometer
PIRD	Project instrumentation requirement document
PIV	Peak inverse voltage
PL	Postlanding
PLSS	Portable life support system
PMP	Premodulation processor
PMR	Pacific Missile Range
PND	Premodulation processor - near earth data
POD	Preflight Operations Division (MSC)
POI	Program of Instruction (NASA)
POL	Petroleum oil and lubricants
POS	Pacific Ocean ship
PP	Partial pressure
PPE	Premodulation processor equipment
PPM	Parts per million
PPS	Pulse per second
PPS	Primary propulsion system
PR	Pulse rate
PRA	Precession axis
PRESS	Pressure
PRF	Pulse repetition frequency
PRM	Pulse-rate modulation
PRN	Pseudo-random noise
PSA	Power and servo assembly
PSA	Power servo amplifier
PSD	Phase sensitive demodulator
PSDF	Propulsion System Development Facility
psia	Pounds per square inch absolute
psig	Pounds per square inch gage
PSK	Phase shift keyed
PSO	Pad safety officer (superseded by PSS)
PSP	Program support plan
PSS	Pad safety supervisor
PTPS	Propellant transfer pressurization system
PTT	Push-to-talk
PTV	Parachute test vehicle
PU	Propellant utilization
PU	Propulsion Unit (NASA)
PUGS	Propellant utilization gauging system
P&VE	Propulsion and Vehicle Engineering (MSFC)
P&VE-ADM	P&VE - Administrative
P&VE-DIR	P&VE - Director
P&VE-E	P&VE - Vehicle engineering
P&VE-F	P&VE - Advanced flight systems
P&VE-M	P&VE - Engineering materials
P&VE-N	P&VE - Nuclear vehicle projects
P&VE-O	P&VE - Engine management
P&VE-P	P&VE - Propulsion and mechanics
P&VE-PC	P&VE - Program coordination
P&VE-REL	P&VE - Reliability

P&VE-S	P&VE - Structures
P&VE-TS	P&VE - Technical and scientific staff
P&VE-V	P&VE - Vehicle systems integration
P&W	Pratt & Whitney
P&WA	Pratt & Whitney Aircraft
PYRO	Pyrotechnic
QA	Quality assurance
QAD	Quality Assurance Division (MSFC)
QAM	Quality assurance manual
QC	Quality control
QD	Quick-disconnect
QRS	Qualification review sheet
QUAL	Quality Assurance Division (MSFC)
QVT	Quality verification testing
RA	Radar altimeter
RAD	Radiation absorbed dose
RAE	Range, azimuth, and elevation
RAPO	Resident Apollo Project Office (NASA)
RASPO	Resident Apollo Spacecraft Project Office (MSC)
RB	Radar beacon
R/B	Radar beacon (RB is preferred)
RBA	Recovery beacon antenna (VHF)
RBE	Radiation biological effectiveness
R/C	Radio command
R/C	Radio control
RCC	Range control center
RCC	Recovery control center
RCC	Rough combustion cutoff
RCS	Reaction control system
RD	Radiation detection
R&D	Research and development
RDMU	Range-drift measuring unit
R/E	Re-entry
REG	Regulator
rem	Roentgen equivalent man
RES	Restraint system
RF	Radio frequency
RFI	Radio frequency interference
RG	Rate gyroscope
RGA	Rate gyro assembly

RGP	Rate gyro package (superseded by RGA)
RGS	Radio guidance system
RH	Relative humidity
RH	Right-hand
RHFEB	Right-hand forward equipment bay
RHSC	Right-hand side console
RJS	Reaction jet system
RO	Reliability Office (MSFC)
RP-1	Rocket propellant No. 1 (kerosene)
RPD	Research Projects Division (MSFC)
RR	Respiration rate
RRS	Restraint release system
RR/T	Rendezvous Radar/ Transponder
R/S	Range safety
RSC	Range safety control
RSC	Range safety command
RSCIE	Remote station communication interface equipment
RSO	Range safety officer
RSS	Reactants supply system
RTC	Real Time Computer (MCC)
RTCC	Real Time Computer Complex (MCC)
RTTV	Real time television
RZ	Return to zero
R&Z	Range and zero
S	ASPO (Apollo Spacecraft Project Office)
S-	Saturn stage (prefix)
SA	RASPO - Atlantic Missile Range
SA	Shaft angle (used by MIT)
SA	Saturn
S/A	Spacecraft adapter
SACTO	Sacramento test operations
SAE	Shaft angle encoder
SAL	San Salvador Island (tracking station)
SAL	Supersonic Aerophysics Laboratory
SAR	RASPO - Atlantic Missile Range (superseded by SA)
SARAH	Search and range homing

SAT	Saturn Systems Office (MSFC)
SBUE	Switch - Backup entry
SBX	S-band transponder
S&C	Stabilization and control
SC	ASPO - CSM (command and service modules)
SC	Signal conditioner (SCR is preferred)
S/C	Spacecraft
SCA	Sequence control area
SCA	Simulation Control Area (MCC)
S/CA	Spacecraft adapter
SCAT	Space communication and tracking
SCATS	Simulation, checkout, and training system (MCC)
SCC	Simulation control center
SCD	Specification control drawing
SCE	Signal conditioning equipment
SCE	ASPO - CSM engineering
SCF	Sequence compatibility firing
SCGSS	Super-critical gas storage system
SCIN	Scimitar notch
SCIP	Self-contained instrument package
SCM	ASPO - CSM (superseded by SC)
SCO	Subcarrier oscillator
S/CO	Spacecraft observer
SCP	ASPO - CSM administration
SCPA	SCS control panel
SCR	Subcontractor
SCR	Silicon controlled rectifier
SCR	Signal conditioner
SCR	RASPO CSM - NAA, Downey
SCRA	RASPO CSM - NAA, Downey - administration
SCRE	RASPO CSM - NAA, Downey - engineering
SCRR	RASPO CSM - NAA, Downey - reliability
SCS	Stabilization and control system
SCT	Scanning telescope
SCT	ASPO - CSM systems test
SCTE	Spacecraft central timing equipment
SCVE	Spacecraft vicinity equipment
SDA	Shaft drive axis
SDF	Single degree of freedom
SDG	ASPO - document control group (obsolete)
SDG	Strap down gyro
SDL	Standard distribution list
SDP	Site data processor
SECS	Sequential events control system
SED	Space Environment Division (MSC)
SEDD	Systems Evaluation and Development Division
SEDR	Service engineering department report
SEF	Space Environmental Facility (NASA)
SEP	Standard electronic package
SEPS	Service module electrical power system
SET	Spacecraft elapsed time
SF	Static-firing
SFX	Sound effects
SG	ASPO - G&C (guidance and control)
SGA	RASPO LEM - GAEC Bethpage (superseded by SLR)
SGC	ASPO - G&C (superseded by SG)
SGE	ASPO - G&C engineering
SGP	ASPO - G&C administration
SGR	RASPO - G&C MIT, Boston
SHA	Sidereal hour angle
SHF	Super high frequency
SI	ASPO - systems integration
SI	Systems integration
S/I	Systems integration (SI is preferred)
S-I	Saturn I first stage
SIA	Systems integration area
S-IB	Saturn IB first stage
S-IC	Saturn V first stage
SID	Space and Information Systems Division (NAA)
S&ID	Space and Information Systems Division (NAA)
S-II	Saturn V second stage

SITE	Spacecraft instrumentation test equipment
S-IV	Saturn I second stage
S-IVB	Saturn IB second stage and Saturn V third stage
S-IVB G	Launch vehicle guidance system
SL	Star line
SL	ASPO - LEM (lunar excursion module)
SLA	Spacecraft LEM adapter
S/L	Space laboratory
SLE	ASPO - LEM engineering
SLM	ASPO - LEM (superseded by SL)
SLM	Spacecraft laboratory module
STLOS	Star line-of-sight
SLP	ASPO - LEM administration
SLR	RASPO LEM - GAEC Bethpage
SLT	ASPO - LEM systems test
SLV	Space launch vehicle
S/M	Service module
SMD	System measuring device
SMJC	Service module jettison controller
S/N	Signal-to-noise ratio
S/N	Serial number
SNA	RASPO NAA, Downey (superseded by SCR)
SNAE	RASPO NAA, Downey engineering (superseded by SCRE)
SNR	Signal-to-noise ratio
S/O	Switchover
SOC	Simulation operation computer (MCC)
SOFAR	Sound fixing and ranging
SOM	Suborbital mission
SOP	Standard operating procedure
SP	ASPO - project integration
SP	Static pressure
SPA	S-band power amplifier
SPACE	Spacecraft prelaunch automatic checkout equipment (superseded by PACE)
SPAF	Simulation processor and formatter (MCC)
SPDT	Single-pole double-throw
SPP	ASPO - program plans and control (superseded by SP)
SPS	Service propulsion system
SPST	Single-pole single-throw
SRD	Spacecraft Research Division (MSC) (superseded by STD)
SRO	Superintendent of range operations
SRS	Simulated remote station (MCC)
SS	ASPO - spacecraft
S/S	Samples per second
S/S	Subsystem
SSA	Space suit assembly
SSB	Single sideband
SSC	Sensor signal conditioner
SSD	Space Systems Division (USAF)
SSDF	Space Science Development Facility (NAA)
SSE	Spacecraft simulation equipment
SSI	ASPO - systems integration (superseded by SI)
SSM	Spacecraft systems monitor
SSO	Saturn Systems Office (MSFC)
SSR	Support Staff Rooms (NASA)
SSS	Simulation study series
SST	Simulated Structural Test (NASA)
SST	Spacecraft Systems test
ST	Shock tunnel
STC	Spacecraft test conductor
STD	Spacecraft Technology Division (MSC)
STMU	Special test and maintenance unit
STS	System trouble survey
STU	Static Test Unit (NASA)
STU	Special test unit

STU	Systems test unit
SVE	Space Vehicle Electronics (DAC) (superseded by Astrionics Branch)
SW	Sea water
SW	RASPO - White Sands Missile Range
SWT	Supersonic wind tunnel
SXT	Space sextent
SYS	System
TACO	Test and checkout station
T/B	Talk back
TBD	To be determined
TC	Test conductor
TC	Transfer control
TC	Transitional control
T/C	Telecommunications
T/C	Thrust chamber
TCA	Thrust chamber assembly
TCA	Transfer control A register
TCB	Technical Coordination Bulletin (MSFC)
TCOA	Translational control A
TCOB	Translational control B
TCSC	Trainer control and simulation computer
TD	Technical Directive (MSFC)
TDA	Trunnion drive axis
TDR	Technical data report
TE	Transearth
TEC	Transearth coast
TFE	Time from event
TJM	Tower jettison motor
TK HTRS	Tank heaters
TLC	Translunar coast
TLI	Translunar injection
TLS	Telescope
TM	Telemetry
TMG	Thermal meteoroid garment
T/M	Telemeter
TP	Test point
TPP	Test point PACE
TPS	Thermal protection system
TR	Test request
T/R	Transmit/receive
TRDA	Three-axis rotational control - direct A
TRDB	Three-axis rotational control - direct B
TRNA	Three-axis rotational control - normal A
TRNB	Three-axis rotational control - normal B
TTE	Time to event
TTESP	Test time-event sequencer plan
TTY	Teletype
TV	Television
TVC	Thrust vector control
TVCS	Thrust vector control system
TWT	Transonic wind tunnel
TWX	Teletype wire transmission
UA	Urinalysis
UDL	Up-data link
UDMH	Unsymmetrical dimethyl hydrazine (fuel)
UHF	Ultra high frequency
USB	Upper sideband
USBE	Unified S-band equipment
ΔV	Velocity change
VAB	Vertical assembly building
VAC	Volts ac
ΔVD	Velocity change display
VDD	Visual display data
VEDS	Vehicle Emergency Detection System (NASA)
VGP	Vehicle ground point
VHAA	Very high altitude abort
VHF	Very high frequency
VLF	Vertical launch facility
VLF	Very low frequency
VOX	Voice-operated relay
VRB	VHF recovery beacon
VSC	Vibration safety cutoff
VTF	Vertical test facility
VTS	Vertical test stand
W/G	Water-glycol
W-G	Water-glycol

WGAI	Working Group Agenda Item (MSFC)	WSMR	White Sands Missile Range
WHS	White Sands, New Mexico (remote site)	XCVR	Transceiver
WIS	Wallops Island Station (NASA) (Wallops, Va.)	XDUCER	Transducer
W/M	Words per minute	XEQ	Execute (PACE)
WMS	Waste management system	XMAS	Extended Mission Apollo Simulation (100 days) (NASA)
WODWNY	Western Office, Downey	XMTR	Transmitter
WOM	Woomera, Australia (remote site)	Z	Astronaut Activities Office
WOO	Western Operations Office (NASA)	ZI	Zone of Interior (continental USA)
WPM	Words per minute	ZZB	Zanzibar, Tanganyika (remote site)

SYMBOLS

ΔP	Delta P	ΔV	Delta V

TERMS

ABLATIVE MATERIAL	During entry of spacecraft into the earth's atmosphere at hypersonic speeds, aids in the dissipation of kinetic energy and prevents excessive heating of the main structure.
ABORT	Premature and abrupt termination of a mission because of existing or imminent degradation of mission success probability.
APOGEE	A point on the orbit of a body which is at the greatest distance from the center of the earth.
APOLLO	A term generally used to describe the NASA manned lunar landing program but specifically used to describe the effort devoted to the development test and operation of the space vehicle for long duration, earth orbit, circumlunar, and lunar landing flights.
APOLLO SPACECRAFT	The vehicle required to perform the Apollo mission after separation of the final launch stage. It consists of the command module (C/M),

APOLLO SPACE-CRAFT (Cont)	the service module (S/M), the lunar excursion module (LEM), the launch escape system (LES), and the spacecraft LEM adapter (SLA).
BACK PACK	A self-contained extravehicular pressure suit life support system.
CIRCUM-LUNAR	Specifically, around the moon. Generally this term has become associated with the program missions in which a spacecraft will circle the moon one or more times and return to earth.
CISLUNAR	Of or pertaining to space between the earth and the orbit of the moon, or to a sphere of space centered on the earth with a radius equal to the distance between the earth and the moon.
CRYOGENIC	An adjective referring to low temperatures, usually those at which gasses become liquid.
DELTA P	Pressure change (ΔP).
DELTA V	Velocity change (ΔV).
DIPLEXER	A device permitting an antenna system to be used simultaneously or separately by two transmitters.
DOCKING	The technique of closing and locking together two or more spacecraft in orbit. The final stage of the rendezvous operation.
EGRESS	The act of or the mechanism for exit from an enclosure. In spacecraft that can relate to the act of a crew member exiting from the vehicle or it
EGRESS (Cont)	can describe the exit chamber, lock and hatchways.
EPHEMERIS	Book of tables giving daily positions of celestial bodies.
HYPER-GOLIC FUEL	Fuel that ignites spontaneously upon contact with an oxidizer, thereby eliminating the need for an ignition system.
INGRESS	The act of or the mechanism for entrance to an enclosure. In spacecraft this can relate to the act of a crew member entering the space vehicle or it can describe the entrance chamber, pressure lock and hatchways.
INTERFACE	The point or area where a relationship exists between two or more parts, systems, programs, persons, or procedures wherein physical and functional compatibility is required.
MICRO-METEOR-OID	Meteoroids less than 1/250th of an inch in diameter.
ORBIT	The path in which one body revolves about another under the gravitational attraction of the latter (as a planet around the sun or a satellite around a planet). The orbit begins and ends at a fixed point in space and requires only 360 degrees of travel.
PERIGEE	That orbital point nearest the earth when the earth is the center of attraction.

REDUNDANCY	The existence of more than one means for accomplishing a given task. Parallel redundancy applies to systems where both means are working at the same time to accomplish the task, and either of the systems is capable of handling the job itself in case of failure of the other system. Standby redundancy applies to a system where there is an alternate means of accomplishing the task that is switched in by a malfunction sensing device when the primary system fails.
RETROGRADE	Slowing a spacecraft or vehicle by applying a thrust in a backward or opposite direction from the direction of motion of the spacecraft or vehicle.
REVOLUTION	Motion of a celestial body or spacecraft in its orbit. A revolution begins and ends at a fixed point or meridian of a planet such as earth. An easterly revolution around the earth is approximately 6 minutes longer than an oribt.
S-BAND	Frequencies in the region of 2500 megacycles per second.
ULLAGE MANEUVER	Forward thrust to the vehicle or stage to shift the propellant to the rear of the tanks prior to firing the S-IVB or SPS engines.
VAN ALLEN BELTS	Two doughnut-shaped belts of high energy charged particles trapped in the earth's magnetic field. The minimum altitude of the inner belt ranges from approximately 100 miles near the magnetic poles to more than 1000 miles at the equator. The maximum altitude of the outer belt extends to approximately 40,000 miles at the equator.